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Report AMCA 70 001

AERIAL VERY HEAVY LIFT CONCEPTS FOR THE 1990 ARMY

VOLUME III - ACADEMIC & INDUSTRIAL PRESENTATIONS

Ad Hoc Working Group No. 6

November 1969

Final Report

U.S. ARMY
ADVANCED MATERIEL CONCEPTS AGENCY

201

Report AMCA-70-001

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VOLUME I - BASIC REPORT

APPENDIX F

INTRODUCTORY REMARKS BY DR. BARNES W. McCORMICK, JR. (CHAIRMAN)

The task of this AHWG is to consider the development of a vertical lift capability for the Army 20 years from now, having a payload of approximately 50 to 60 tons. If the discussion seems to be somewhat biased in favor of helicopters, it is hoped that the representatives from Goodyear, LTV, and others, will bear with me because, off hand, you would agree that as of now this is the obvious way to go. Formulated as a result of this meeting is a report with recommendations as to the problem areas involved with the development of a very heavy lift vehicle. It does not take too much thought to develop a formidable list and I am sure that a lot of you, particularly from industry, already have such a list. For example, we have the problem of transmissions. The transmission must be large enough to transmit the power required and must have a high enough gear reduction to maintain a reasonable tip speed. There is the problem of satisfactorily predicting the structural weight for aircraft or helicopters of this size. The general area of vibrations will present a problem since the natural frequencies of components will go down. Ground and air resonance will need to be given some attention.

The mission should be considered before continuing further. What will be required of the vehicle in terms of speed and range? What will it be expected to do? If we are considering helicopters, what type of rotor should it have; should it be a single rotor, a tandem or a multi-rotor? Are we considering only one helicopter? Maybe we should consider coupling several vehicles together. We should give attention to the handling problem with external loads. Of course, most of these areas have been treated in the heavy lift studies that have been done to date but still we are talking about a significant increase in the payload so that these problem areas will have to be re-examined.

Noise is another important consideration. How does noise vary with rotor size for a constant disc loading and tip speed? Might the noise not be prohibitive for a very heavy lift helicopter. The present QMR for the heavy lift helicopter contains some noise requirements. Without recalling the specific numbers, it does refer to the avoidance of large impulsive noise levels in the cabin. It is also believed that there are some requirements listed (in the QMR) regarding the far-field noise.

A few very preliminary calculations and observations will be presented which have been made based on the extrapolation of today's technology. It is realized that these may be a little naive by comparison with the design studies which some of you from industry might have made.

The first graph (Figure 1) of disc loading versus the one-third power of the gross weight illustrates what happens in the scaling of helicopters. We have all heard about the square-cube law and more will be heard later concerning scaling. This figure illustrates the application of this law to helicopters. To explain this graph, suppose a helicopter is simply scaled geometrically to larger and larger sizes. The weight would be proportional to the cube of some characteristic length while the disc area would vary with the length squared. The disc loading would thus be proportional to the one-third power of the weight. As you can see the data shown here does indeed fall along a straight line. Represented here are tandem rotors, Russian helicopters and single rotor helicopters.

The next figure (Figure 2) presents the empty weight as a function of normal gross weight for helicopters. It is somewhat surprising that the data lies along a straight line with the exception of the crane configurations which one would expect to have with a relatively lower empty weight. Generally, the empty weight is approximately 60% of the gross weight even for the largest helicopter. This is somewhat difficult to understand on the basis of the square-cube law. If the same stress levels are maintained for all designs, one would expect the structural weight as a percentage of the gross weight to increase with the gross weight.

Figure 3 presents a look at power loading. This looks somewhat like the "shot gun" pattern of data which one tries to avoid. Ideally, of course, the power loading, pounds per horsepower, should vary with the reciprocal of the disc loading. Hence plotted is the available data in the manner shown and includes lines representing constant figures of merit. As used here, this figure is the ratio of the ideal power required to the installed shaft horsepower. No attempt was made to separate out the rotor power. The power used was the total installed power and hence includes margins for engine-out performance, etc. Generally, as you can see the data lies between figures of merit from .385 to .605.

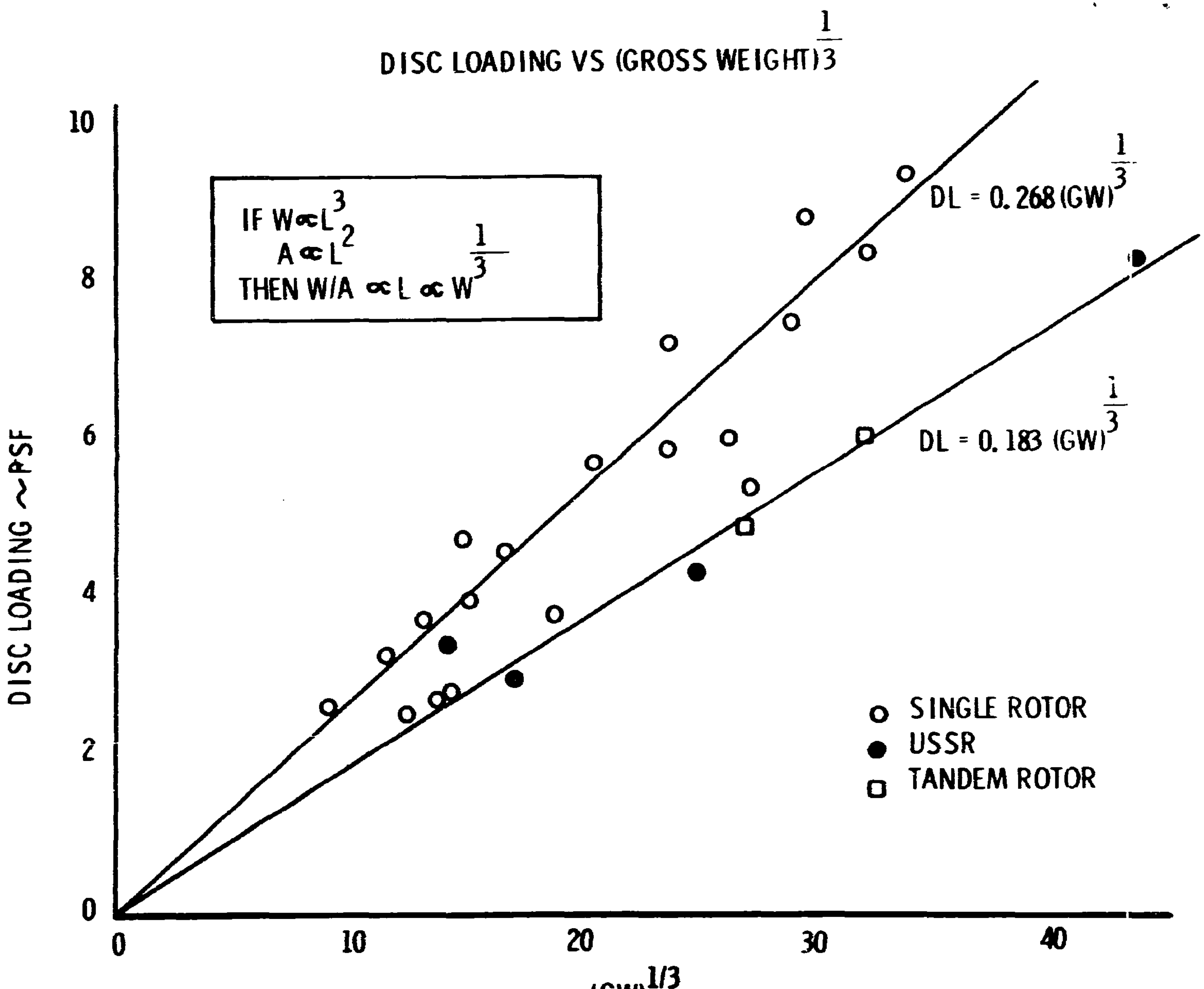
In order to size the very heavy lift helicopter, it is also good to take a look at the rotor average lift coefficient, $\overline{C}_L$. The data for $\overline{C}_L$ as a function of gross weight is shown in Figure 4. There seems to be a tendency for $\overline{C}_L$ to decrease somewhat at the higher weights. It is difficult to understand why this should be and within the scatter of the data it is not definite that any significance should be drawn from it.

A typical value of $\bar{C}_L$ of 0.5 should be satisfactory.

Using the data from the foregoing figures, a little "guesstimating" has been done as to what a very heavy lift helicopter might look like. Chosen was a $\bar{C}_L$ of .45 and a tip speed of 700 feet per second. The weight empty was assumed to be .62 of the gross weight and the disc loading to be in line with that of the large Russian helicopters shown in the first figure. A figure of merit of 0.5 was chosen based on the installed power and a brake specific fuel consumption of 0.6 gals. per b.hp./hr. Further assumed was an equivalent flat plate area of 400 sq. ft. for the helicopter and 150 sq. ft. for the external payload. The QMR on the heavy lift helicopter specified 100 sq. ft. of flat plate area for the bulky payload which, using the square-cube law, was scaled up to 150 square feet for this payload.

Using these assumptions and going through several iterations, a fuel weight of approximately 46,000 pounds was calculated. Obviously, to do this some sort of a mission had to be defined. Chosen was the same as the one currently specified for the heavy lift helicopter, that is, a 25 mi. radius, two round trips, 10 minutes of hovering with payload and 20 minutes of hovering without. A 50 ton payload was assumed for the calculations. The disc loading which resulted from the calculations was 13.2 psf and the gross weight was 384,000 pounds. The calculated installed horsepower was 73,400 shp; the rotor diameter was calculated to be 192 ft with a solidity of .151. For example, if the rotor had six blades, the chord of each blade would be 7.6 ft.

These preliminary numbers are presented simply as a starting point with the realization that more detailed study must be done. (Note: the ensuing discussions by the committee suggested that the preceding numbers were too pessimistic and that the 50-ton payload helicopter would probably be lighter than the numbers herein indicated).



DISC LOADING VS (GROSS WEIGHT) 1/3
IF W ∝ L^3
A ∝ L^2
THEN W/A ∝ L ∝ W^1/3
DL = 0.268 (GW)^1/3
DL = 0.183 (GW)^1/3
SINGLE ROTOR
USSR
TANDEM ROTOR
DISC LOADING ~ PSF
(GW)^1/3
0
2
4
6
8
10
0
10
20
30
40

FIGURE 1

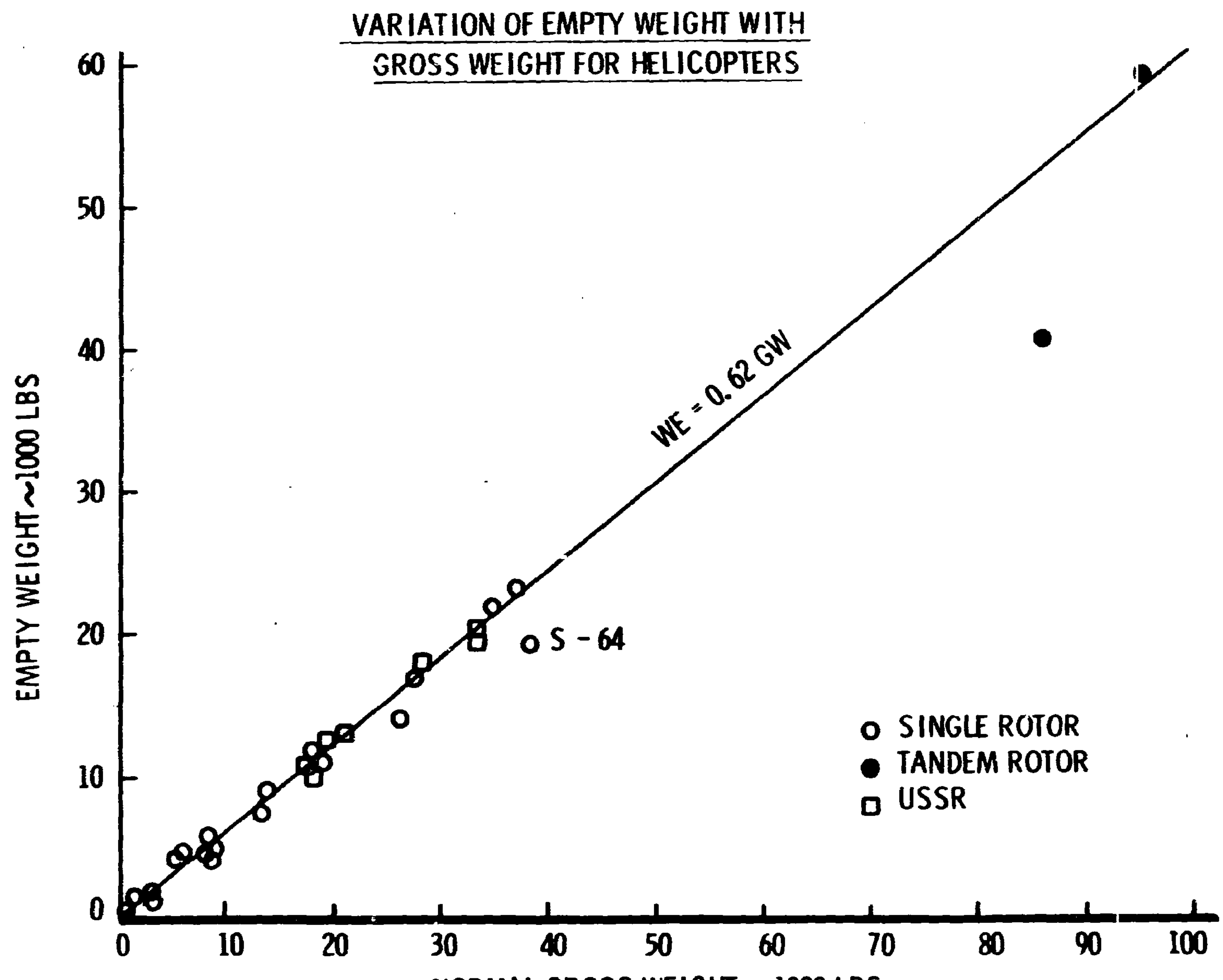
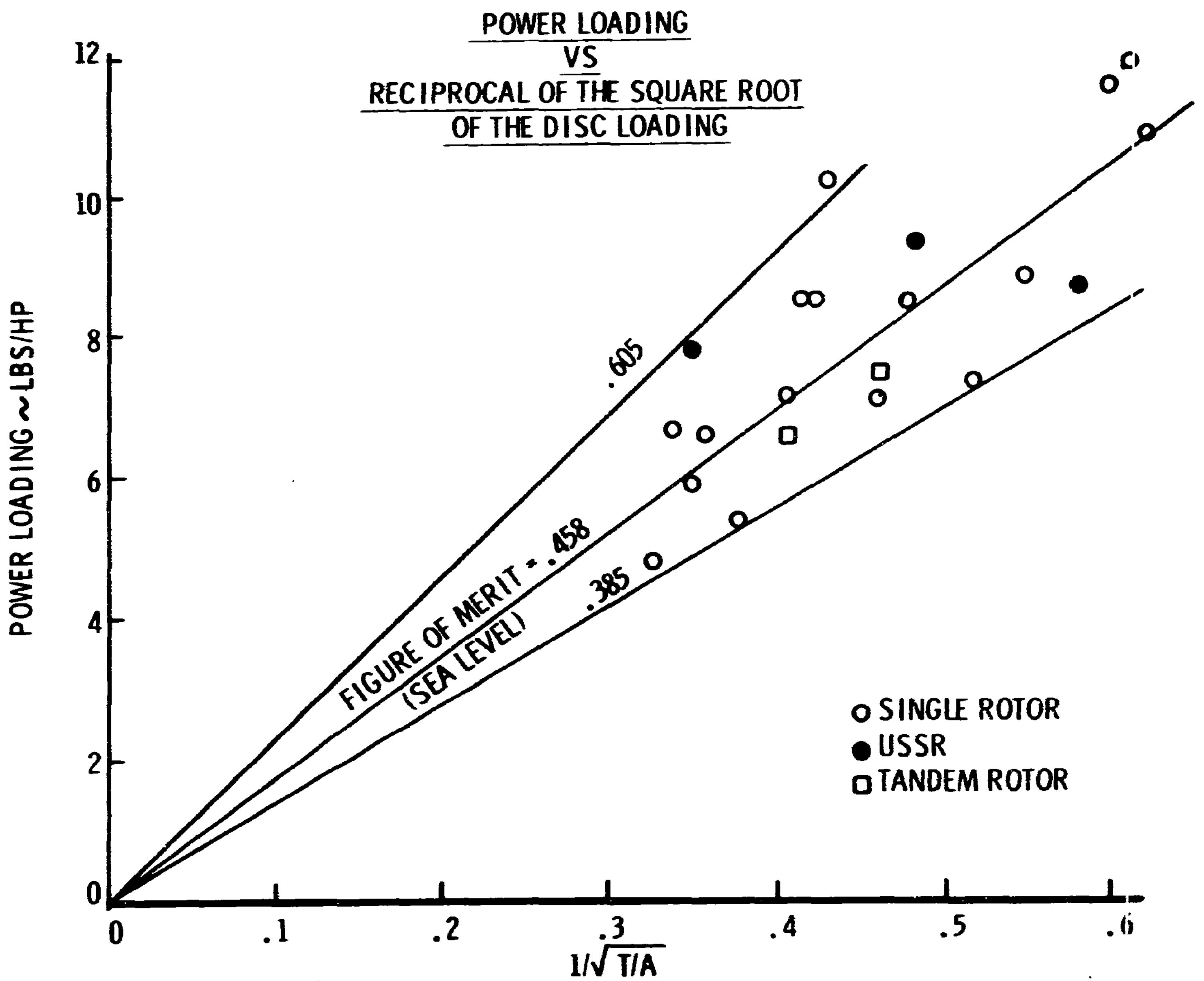
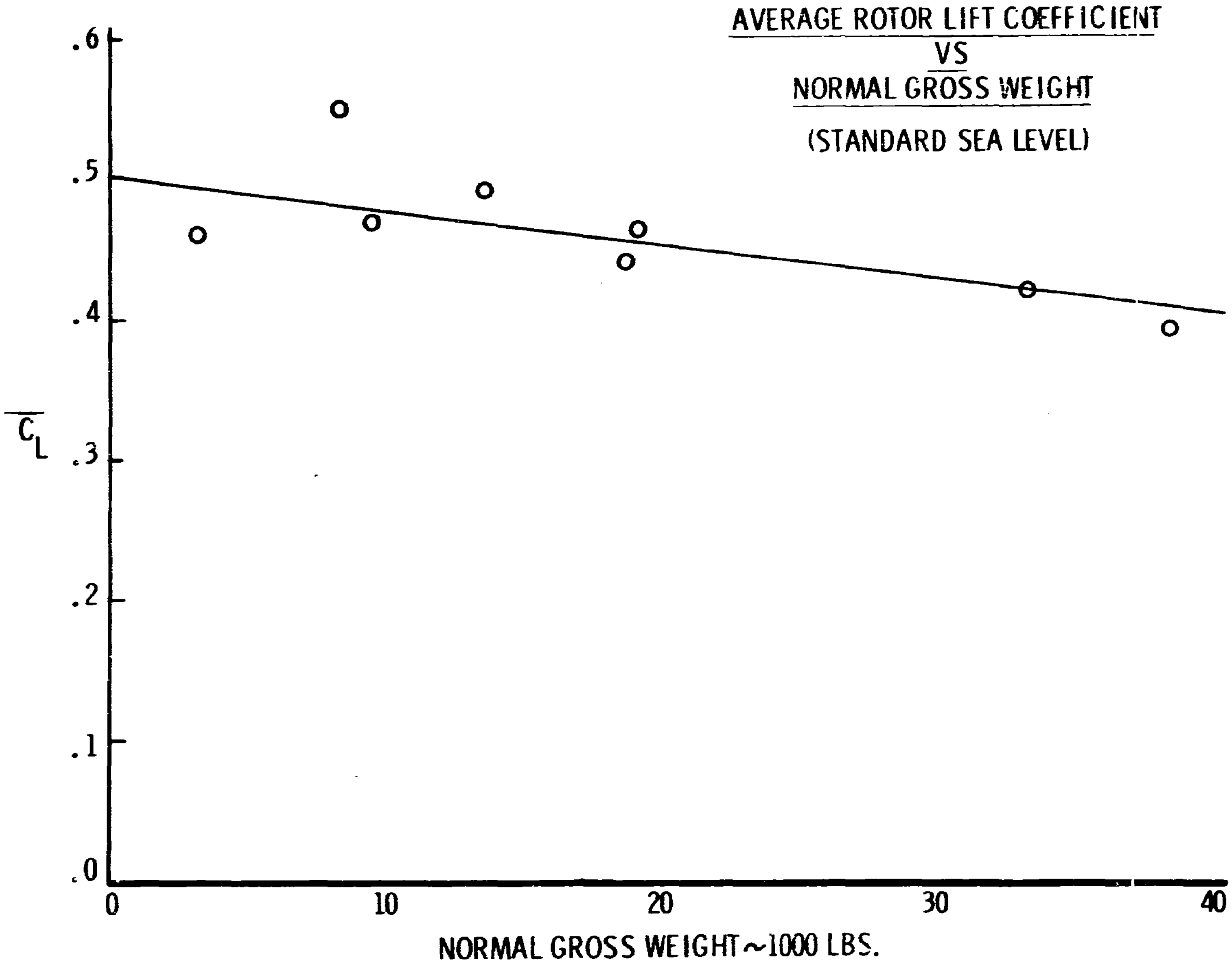

VARIATION OF EMPTY WEIGHT WITH
GROSS WEIGHT FOR HELICOPTERS
EMPTY WEIGHT ~1000 LBS
NORMAL GROSS WEIGHT ~1000 LBS.
WE = 0.62 GW
S - 64
SINGLE ROTOR
TANDEM ROTOR
USSR
0
10
20
30
40
50
60
70
80
90
100

FIGURE 2



POWER LOADING
VS
RECIPROCAL OF THE SQUARE ROOT
OF THE DISC LOADING
POWER LOADING ~ LBS/HP
12
10
8
6
4
2
0
0
.1
.2
.3
.4
.5
.6
$1/\sqrt{T/A}$
FIGURE OF MERIT = .458
(SEA LEVEL)
.605
.385
SINGLE ROTOR
USSR
TANDEM ROTOR

FIGURE 3



AVERAGE ROTOR LIFT COEFFICIENT
VS
NORMAL GROSS WEIGHT
(STANDARD SEA LEVEL)
$\overline{C_L}$
.6
.5
.4
.3
.2
.1
.0
0
10
20
30
40
NORMAL GROSS WEIGHT ~1000 LBS.

FIGURE 4

APPENDIX G
GENERAL DESIGN CONSIDERATIONS

Presentation by Dr. George Wislicenus - Penn State University

Simply specifying the payload which we want to lift vertically is not a sufficient criterion in design of a VHLH vehicle. Certainly the hovering time must be specified; after all, the rocket engine that just launched our astronauts to the moon produced a thrust of 7.5 million pounds, but not for a very long time. So we have to ask, "How much time, for what distance, and at what speed?" If distance and speed are important it is believed that something other than a helicopter, probably along the lines of a VTOL aircraft, must receive consideration. In addition to the weight of the payload the bulk of the payload should also be considered. If it is very bulky its aerodynamic drag during horizontal flight may become prohibited. However, it is an interesting idea that the type of aircraft we are considering could conceivably transport a load more bulkier than could be transported over land by any other means.

I would now like to discuss briefly a study conducted in cooperation with a graduate student of mine where we considered the problem of scaling larger and larger aircraft. This study was not concerned with V/STOL aircraft and helicopters per se. We wanted to understand why it has apparently been proposed to build larger aircraft which are not in accordance with the well-known square-cube law. In passing, it should be noted that lighter-than-air aircraft have no problem in this respect since they derive their lift by buoyancy rather than by dynamic means.

Since we are dealing with gravitational and dynamic forces the Froude number is an obvious criterion. Figure 1 presents the reciprocal of the cube root of the Froude number as a function of the take-off weight of aircraft. As defined here the Froude number is the ratio of gravitational forces to the dynamic forces. It is interesting that this number is nearly constant for propeller driven aircraft as a result of the speed increasing with weight. In the subsonic jet field, however, this is not the case because these aircraft are flown more or less at the same speed, close to the sonic velocity, regardless of size. This latter case is probably comparable to what we would find with helicopters where the tip velocity of rotors, or similarly for lift fans, is more or less constant.

Figure 2 presents the ratio of structural weight to gross weight for fixed wing aircraft as a function of the gross weight. As you can see, the ratio varies very little as a function of gross weight and, if anything, decreases slightly with increasing size. According to the square-cube law, this should have been in the opposite direction. This is strong evidence that we are "beating" the square-cube law. Incidentally, the figure which Dr. McCormick presented for helicopters was about 0.6. The Russian helicopters were approximately .48 which is doing pretty well by comparison with this figure.

The area loading for fixed wing aircraft is presented in Figure 3. The reference area which was used was the total vertical projected area of the aircraft. This, of course, is more or less proportional to the wing area but it was used since the fuselage may also provide some lift. This figure implies that as we progress to larger aircraft the area is not increasing as fast as the weight. This is one way we circumvent the square-cube law in the growth of aircraft. Interestingly enough, as a sideline, the ratio of the weight to the area to the three halves power (which is in effect the volume) is nearly constant for propeller-driven aircraft and decreases slightly for jet aircraft with increasing size. In other words, the density of these aircraft diminishes slightly with increasing gross weight.

Figure 4 presents the wing lift coefficient as a function of maximum take-off weight. This lift coefficient is calculated for cruising conditions and shows a tendency to increase with increasing size which once again reflects the higher wing loading.

Figure 5 addresses itself more specifically to the problem of negating the square-cube law with regard to fixed wing aircraft by refinement of the structure. Here an average skin thickness, σ, is divided by the square root of the vertical projected area. C_W is simply a constant describing the weight of the aircraft in terms of its area and thickness and is equal to the weight divided by the material density, the area, the average skin thickness and the acceleration of gravity. This graph shows that the average skin thickness in proportion to the linear dimension of an aircraft has remained nearly constant for propeller-driven aircraft as the sizes have increased. However, in the case of subsonic jets this ratio has decreased with increasing size pointing to continual improvement in the structural design.

Figure 6 is perhaps the most revealing relative to the upgrading of the quality of the mechanical design. The constant, C_b, is a dimensionless measure of bending stress at the wing root and is equal to the moment divided by the root section modulus. The larger C_b the better the aircraft has been designed with respect to bending stress. The dimensionless coefficient, C_m, is equal to the wing bending moment divided by the weight of the aircraft and a characteristic length. The smaller this coefficient the better is the distribution of the wing loading to minimize the bending moment. To emphasize again the larger C_b and the smaller C_m the better will be the design. This figure then presents the product of the material stress and C_b divided by the product of C_w and C_m. The larger this combination the better will be the structural design of the aircraft. As you can see, this combination of parameters has increased by a factor of approximately 4 in going from a gross weight of approximately 20,000 to 800,000 pounds. The aircraft represented here are constructed primarily of aluminum and we know that the strength of the material has not increased by a factor of four. We must conclude that the quality of the mechanical design has been improved considerably over the years with the growth of fixed wing aircraft.

A similar combination of parameters is presented in Figure 7. Here C_σ is equal to the product of the material density and the square root of the weight is divided by the stress to the three-halves power. The combination of constants shown here represents a similarity criterion with regard to weight and strength. If this coefficient is a constant, one can satisfy similar operating conditions with similar structures. From this figure, within the scatter of the data, it can be seen that this combination of parameters is essentially constant as a function of gross weight.

The next four figures illustrate some results of a design problem which were done a few years ago relating to a design class. Considered was the possibility of designing a VTOL aircraft to lift a bulky load of 100,000 pounds. I had in mind a civil application, namely to deliver factory built houses from the factory directly to the site. It was assumed that the house would be 70 feet wide, 25 feet high, and 80 feet long. This is a bulky load; something which could be transported over land by conventional means. Please understand that these sketches are strickly preliminary in nature.

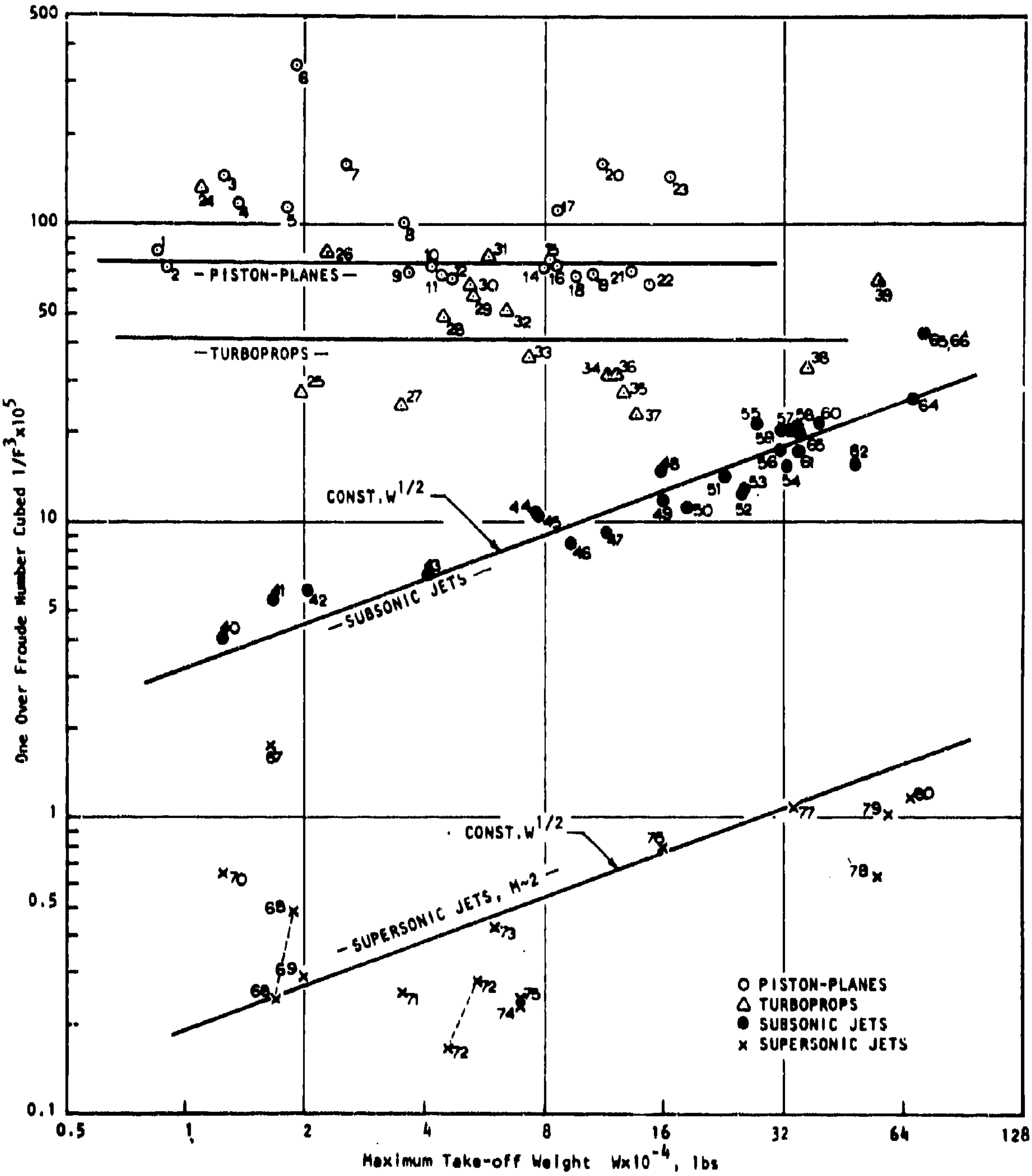
Figure 8 shows a multilift fan aircraft which was considered first. As you can see, the lift fans are very large. The payload area is in the center and is not to be crossed with any beams. The way in which the structural members would have to run are shown diagrammatically. One of the problems is the fact that if you have a very large area occupied by the payload then the rest of the aircraft by necessity becomes large also. This is feasible only if your aircraft has less weight per unit area than present day aircraft.

Figure 9 is a side view of the aircraft illustrating louvers in the fan which are adjustable to provide forward thrust for the horizontal fixed wing mode of flight.

Figure 10 may cause the helicopter experts present to shutter but it will be presented nevertheless. In this planform view of the preliminary design it can be seen that the rotor hub was made sufficiently large in order to cover the payload. This avoids hanging the very bulky load in the downwash from the rotor.

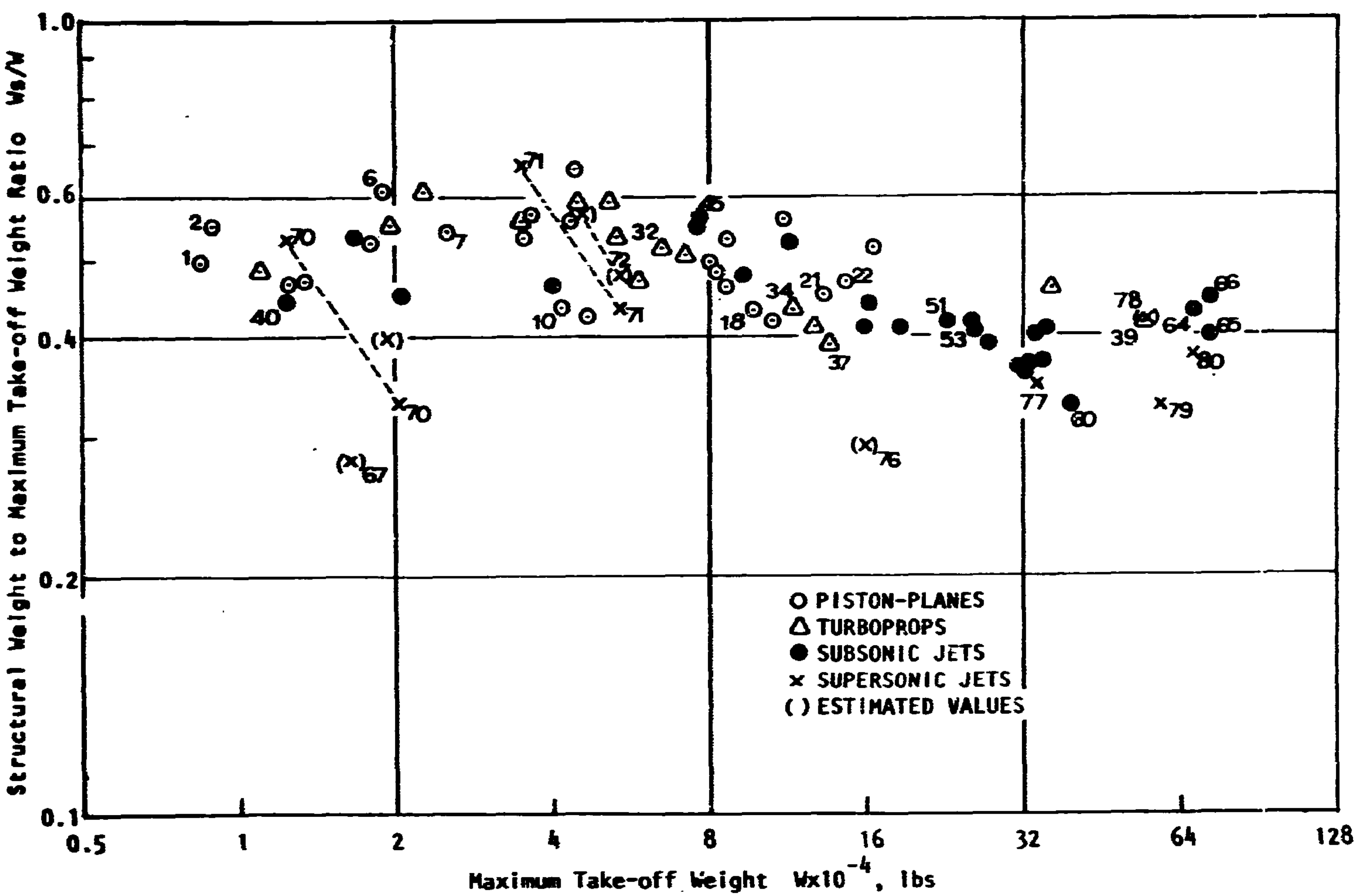
Figure 11 is a side view of the helicopter. The blades are shown relatively large as I had in mind some sort of a jet reaction drive such as the warm cycle.

Thus, in conclusion, it is entirely clear that a multiplicity of refinements in the structural design of fixed wing aircraft has accomplished what we accept today. The existence of fixed wing aircraft of the order of 800,000 pounds gross weight is a matter of good design, and I have not the remotest doubt but that we can build a VTOL aircraft to lift 50 or 60 tons of payload. It is hoped that we will not limit ourselves in our ensuring discussions solely to the helicopter. If we require very long ranges with reasonable efficiency, then the lift fan should also be given attention.



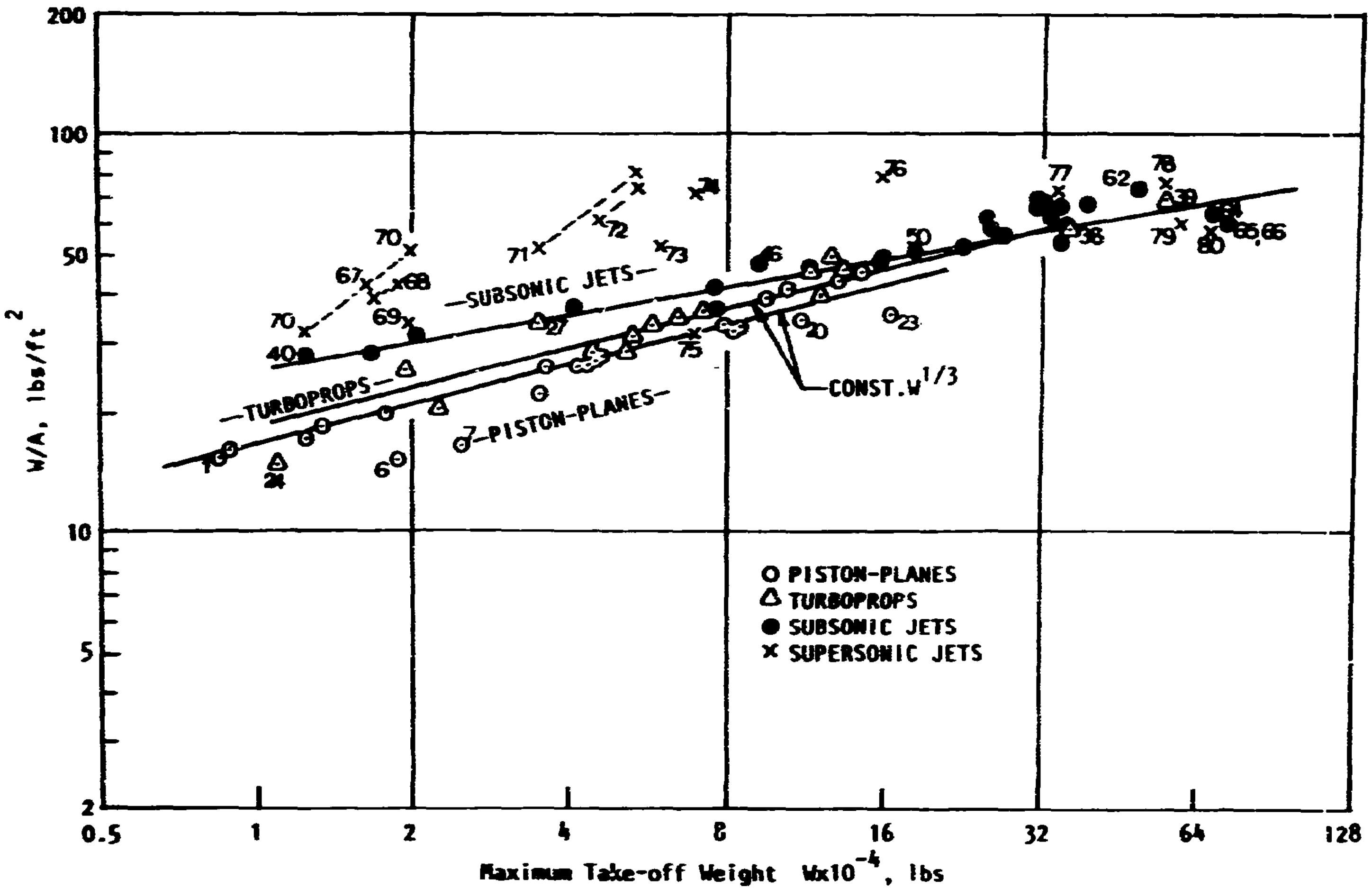
One Over the Froude Number Cubed.

FIGURE 1



Structural Weight to Maximum Take-off Weight Ratio.

FIGURE 2



Maximum Take-off Weight to Characteristic Area.

FIGURE 3

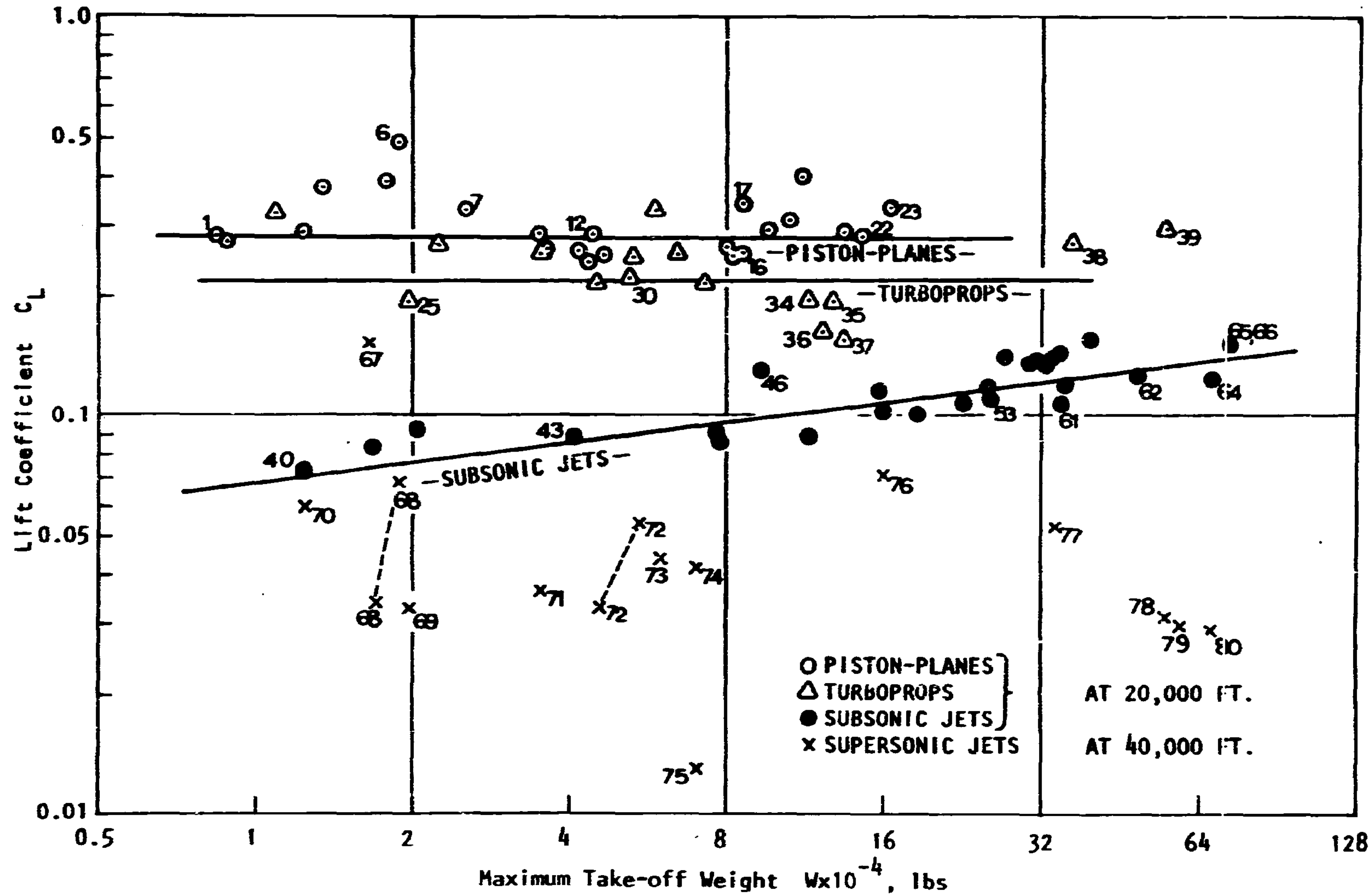
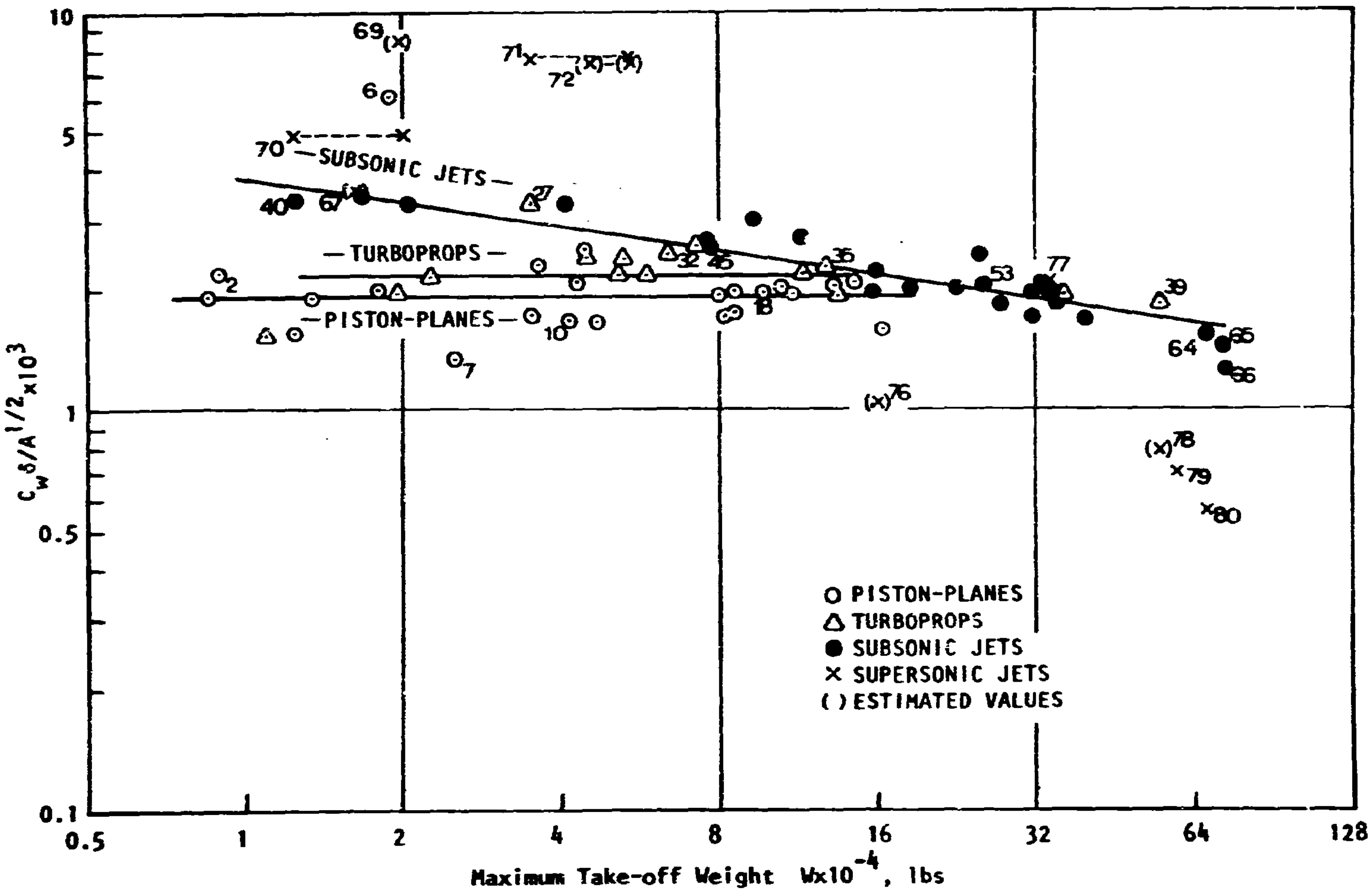


FIGURE 4

Lift Coefficient



$C_w \delta / A^{1/2}$.

FIGURE 5

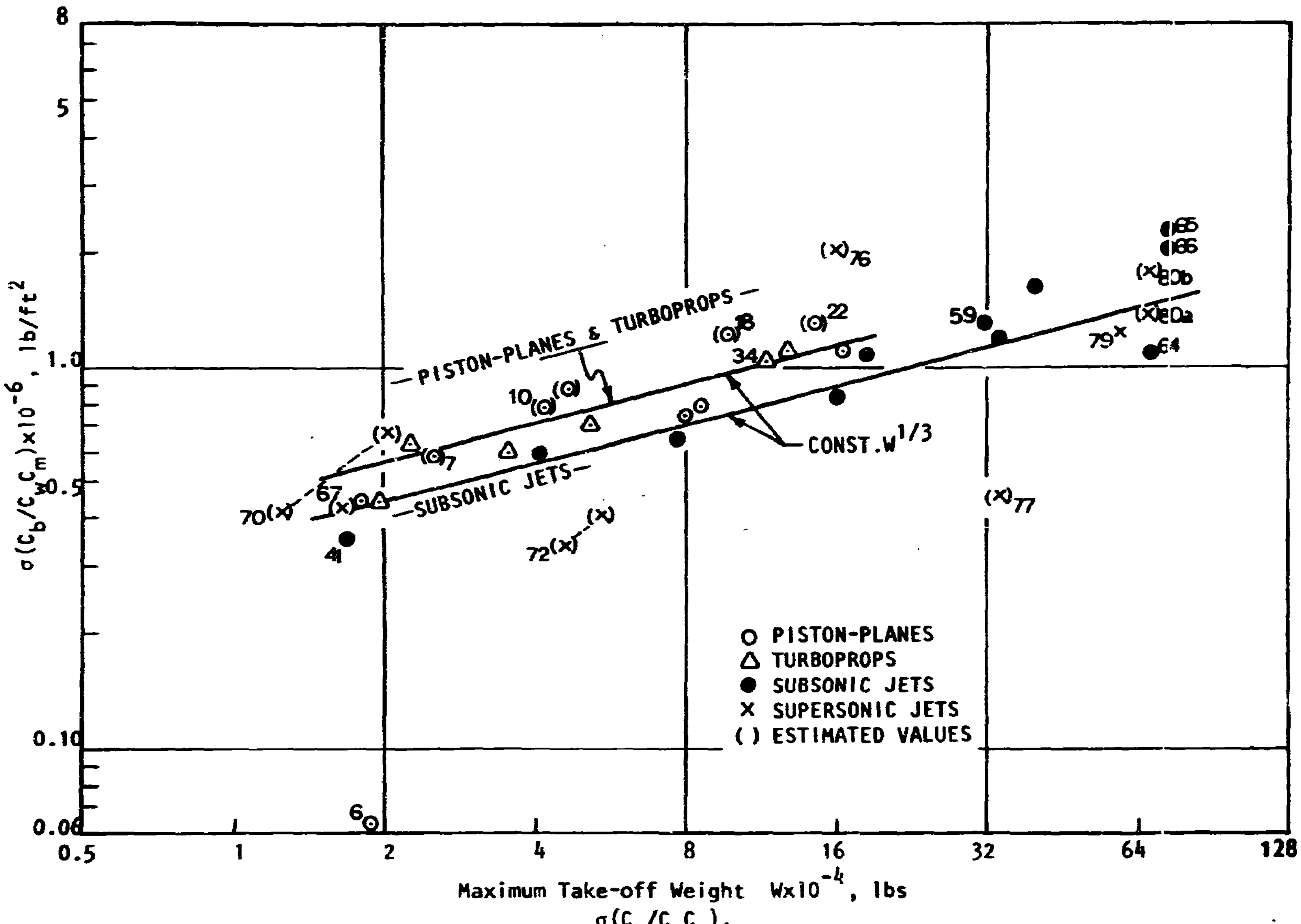
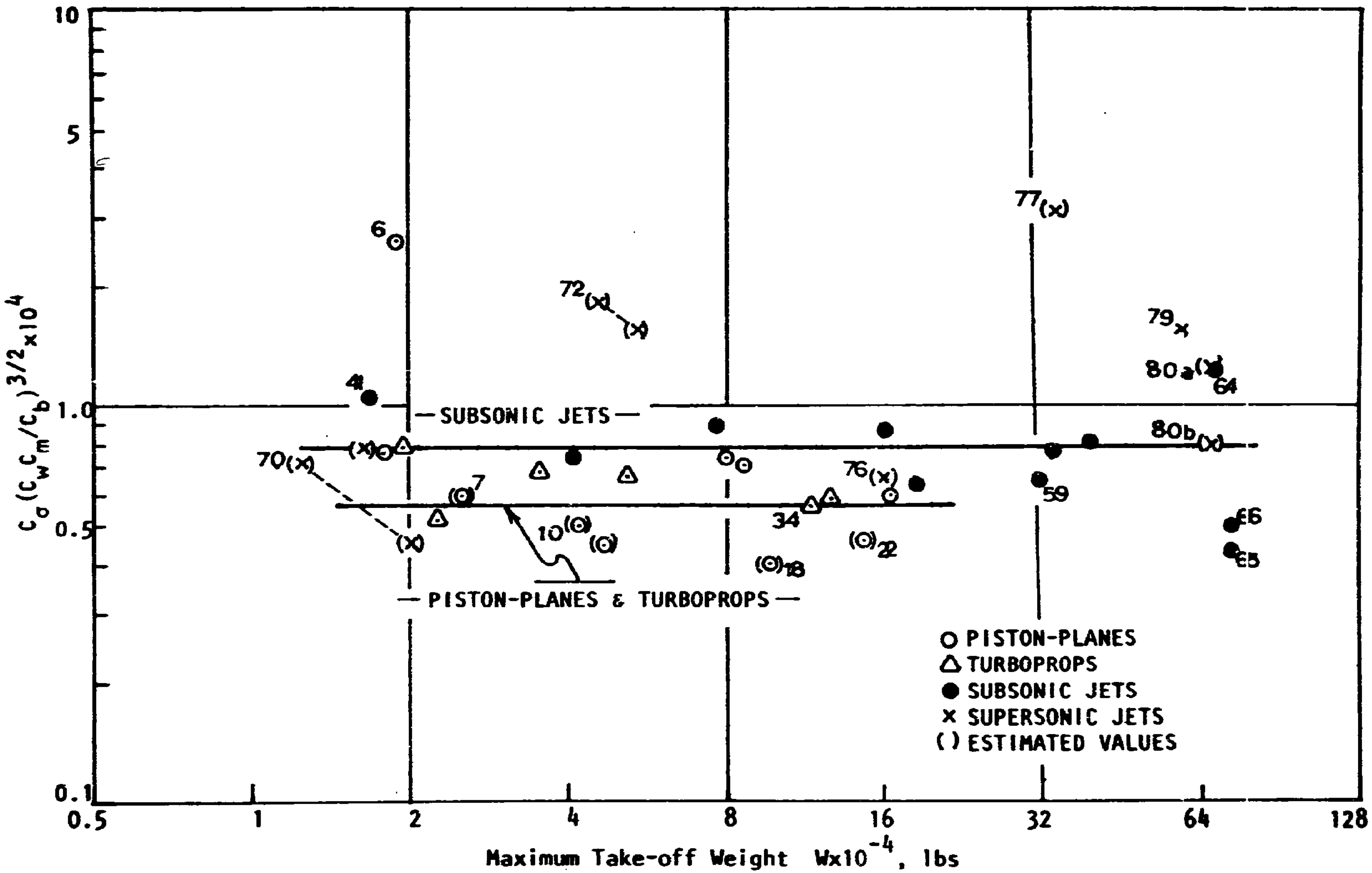

PISTON-PLANES & TURBOPROPS
SUBSONIC JETS
CONST.$W^{1/3}$
○ PISTON-PLANES
△ TURBOPROPS
● SUBSONIC JETS
X SUPERSONIC JETS
( ) ESTIMATED VALUES
$\sigma(C_b/C_wC_m)\times10^{-6}$, lb/ft^2
Maximum Take-off Weight $W\times10^{-4}$, lbs
$\sigma(C_b/C_wC_m)$.

FIGURE 6



$C_\sigma(C_wC_m/C_b)^{3/2}$.

FIGURE 7

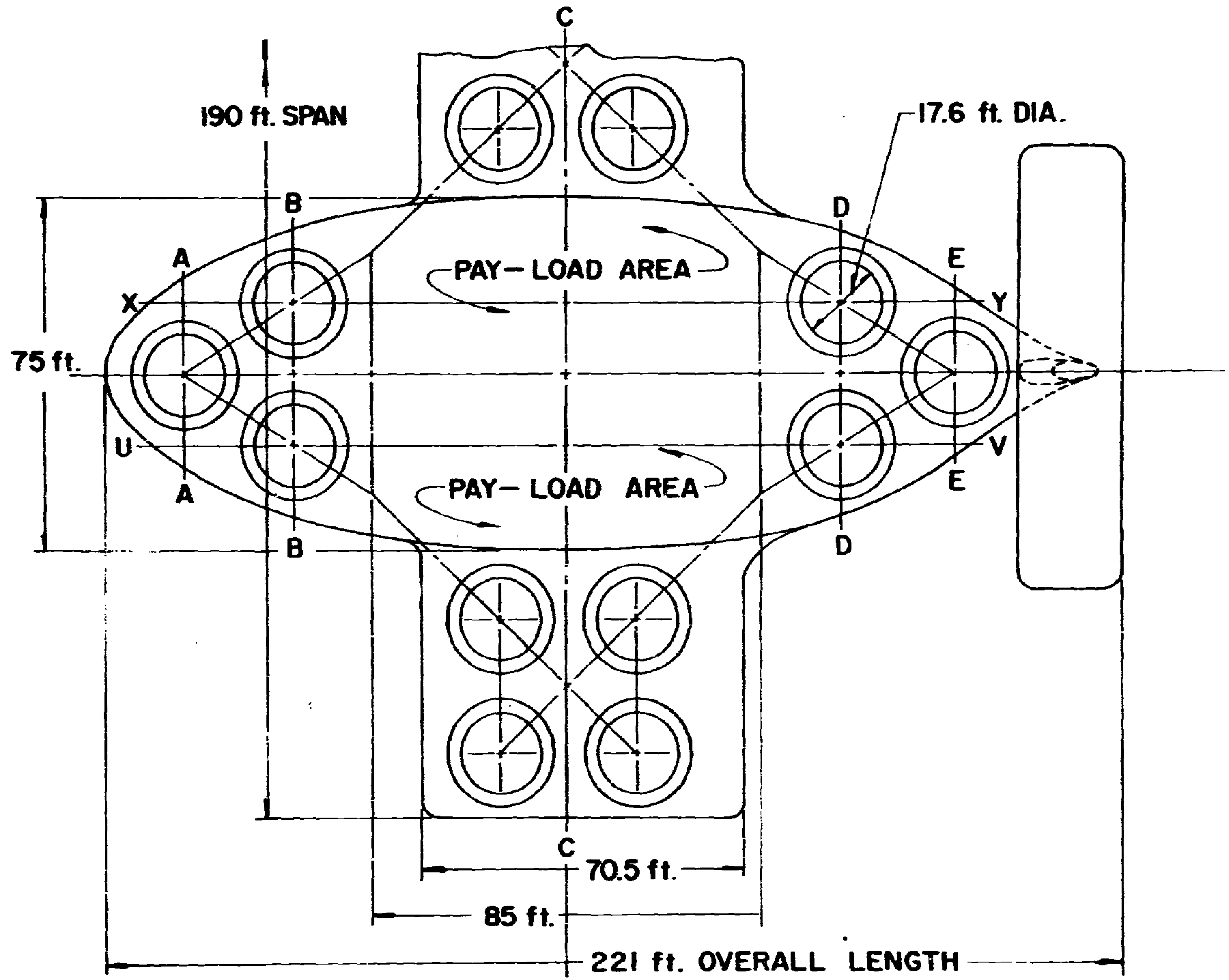

190 ft. SPAN
17.6 ft. DIA.
C
B
D
A
E
X
Y
75 ft.
U
V
PAY— LOAD AREA
PAY— LOAD AREA
70.5 ft.
85 ft.
221 ft. OVERALL LENGTH

FIGURE 8

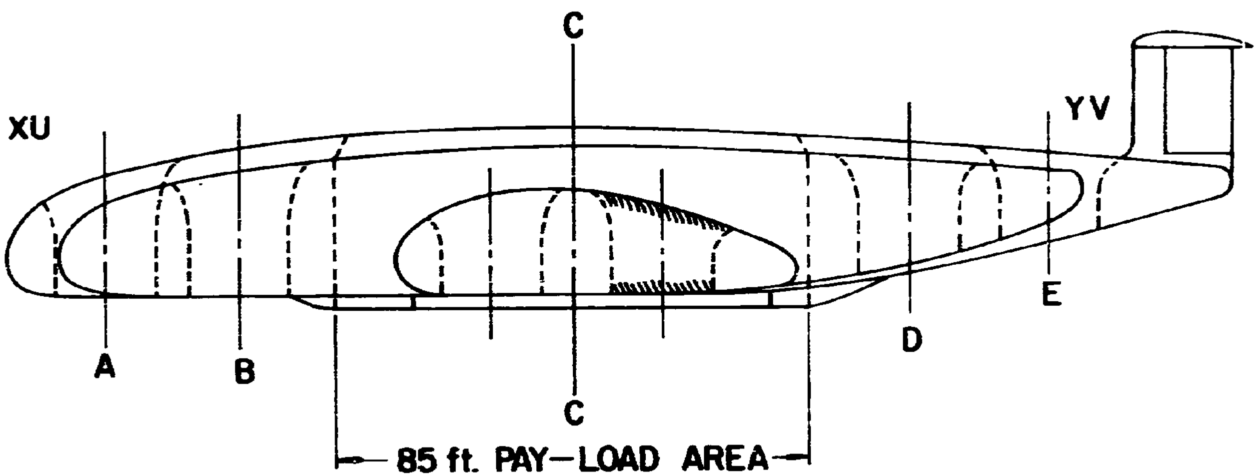
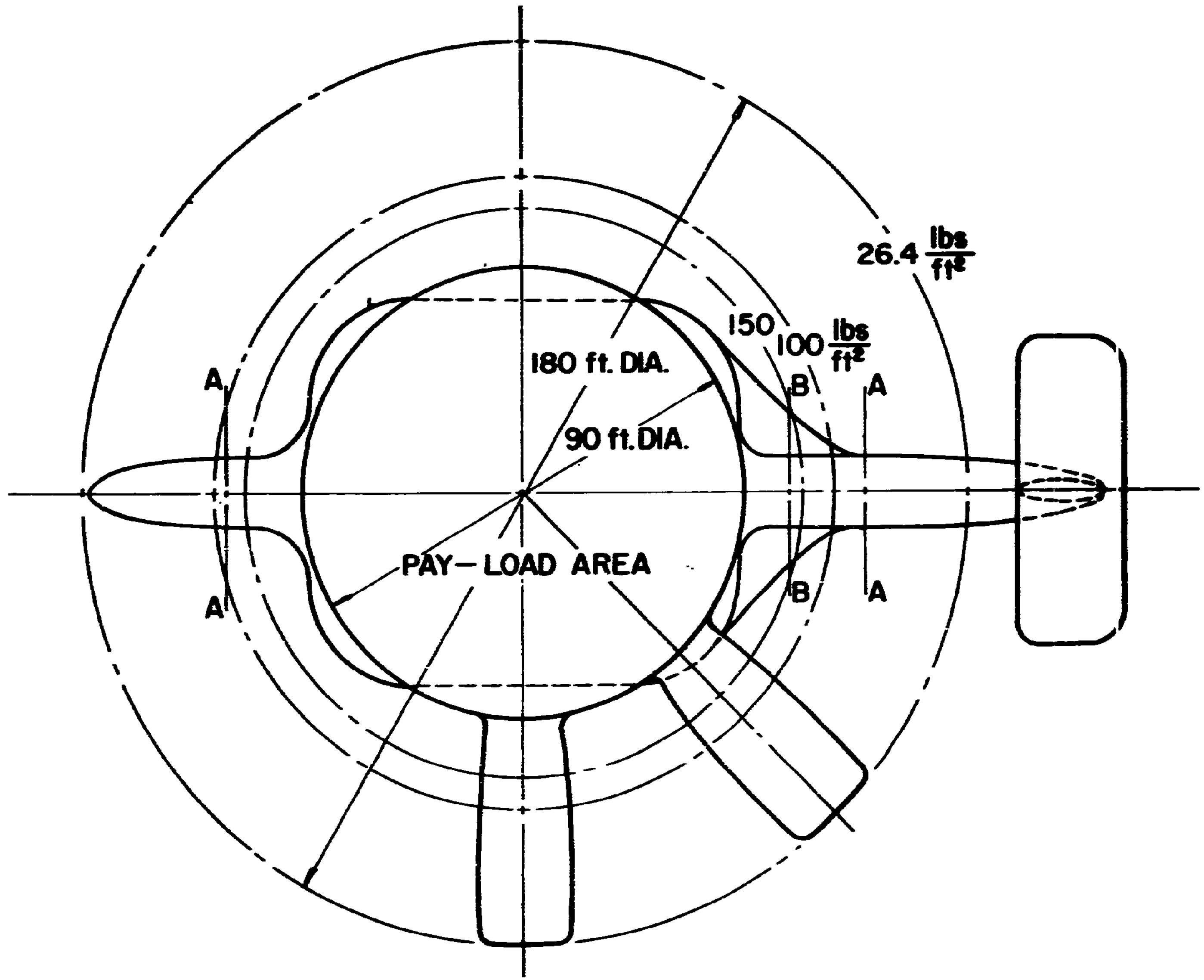


FIGURE 9



26.4 lbs/ft²
150
100 lbs/ft²
180 ft. DIA.
90 ft. DIA.
PAY—LOAD AREA
A
A
A
A
B
B

FIGURE 10

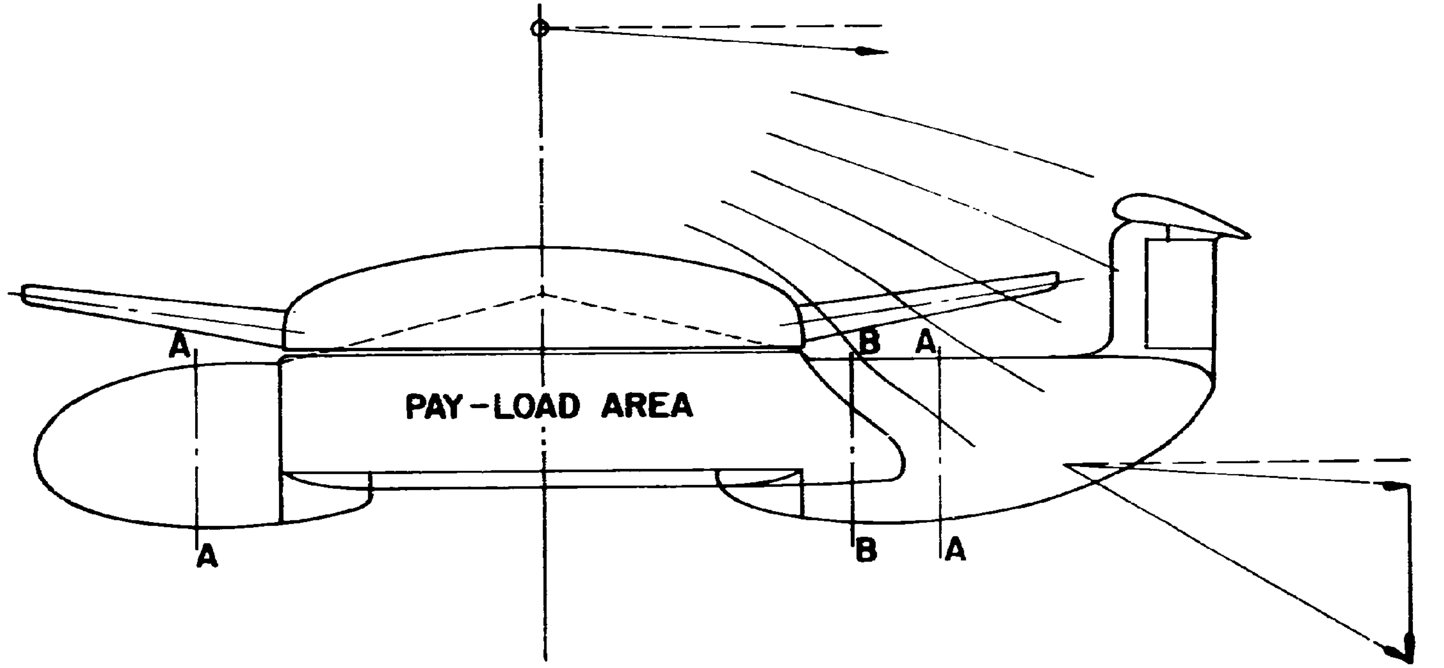


FIGURE 11

APPENDIX H
FUTURE PROPULSION TECHNOLOGY

Presentation by Mr. Lester Veno-General Electric

Thus far discussions have been concerned with requirements, and helicopter design considerations. There have been several remarks made with reference to Propulsion Systems and naturally in our business we feel that the Propulsion System may be a very important part of the VHLH, but perhaps, not the limiting part. I will agree with some of my competitors here that we don't feel that the propulsion system or the propulsion technology will be limiting.

Figure 1 implies that we are going to talk about turbo-shaft engines and I think I should correct one impression right away and that is, that although this discussion will be confined to turbo-shaft engines, we in GE, feel that a turbo fan engine may also be a propulsion system for a VHLH as a gas supplier. However, this discussion and the technology that we will cover here will be mostly confined to turbo-shafts.

One of the first things that we must consider is the market need (Figure 2). After all we are in business for profit and we like to look at the market and to try to do our planning accordingly, and as we observe the whole market at the moment we find that we can categorize it or classify it in each of five groups: The 400 to 800 horsepower level, is shown first and we list some typical applications for this size engine. We then go to the next class between 1,000 and 2,500 horsepower and the third size, 4,000 to 6,000. The fourth size is 6,000 to 10,000. And a final larger group which we will focus our attention on today which would be 8,000 horsepower and up. I have listed some of the potential applications shown covering all branches of the military service. Naturally we would like to design an engine that would have a multiple of uses. If you look at GE's spectrum of engines at the moment (Figure 3) you find that we have two production engines shown in the lower left hand side. The T58 engine running from an initial start of about 1,000 horsepower to 1,500 horsepower, and the T64. At the end of these particular little arrows is shown the date of the initial MQT, and the growth and the width of the arrow, as shown here, is the relative growth in the horsepower of the engine during its life history. I show four new engines, which we call advanced technology engines. We see an engine up at the top labeled "A", and I will identify that as a shaft derivative of our TF 39 engine, the large high bypass ratio turbo fan used in the C5A. We feel that a 25,000 horsepower engine could be built very easily from that particular configuration. It is a high temperature cycle. I am going to have a little more to say about that later, but you can see that we believe that there are engines available, or could be available at 25,000 horsepower up to nearly 30,000 horsepower. The category called "B" is a group of engines that are derived from our J97 Air Force-funded engine. This is a 65 pound per second gas generator and has a considerable latitude for growth as you can see from about 12,000 shaft horsepower up to 15,000 or higher. The horsepower values I am quoting you are on a standard day. There is a family in "C" based on our Navy developed TF34 engine. This is an engine for the VSX. It is a family derivative of the T64. It has a capability of between 6,000 and 8,000 horsepower. Finally there is a family "D", shown at the lower end of the size scale. This is the Army-supported GE-12 engine which is being developed for UTTAS. It has the potential of between 1,500 and 3,000 horsepower, depending on what size is finally configured. You can see its timing is in the period from 1974 through 1976. I think our interests today are in the upper two, the larger engines, A and B. We also identified a broad band here where we feel advanced technology engines could be developed.

Now, how is technology manifested in engine designs. There are two ways here of looking at it. We have plotted in Figure 4 in very broad terms two basic engine design parameters, the cycle pressure ratio and turbine inlet temperature versus years. You note that our production engine, T58 and T64 indicate a steadily increasing trend to higher cycle pressure ratio, then we find our advanced engines, A, B, C, and D, tend to fall even higher up at the upper end of the scale. The turbine inlet temperature trend has a jog in it which was introduced not too many years ago around 1965 when we went from uncooled turbine blades to cooled blades, and I think that most of you are aware that there is a great deal of increased sophistication being factored into methods of cooling turbine blades to allow even higher turbine inlet temperatures.

Now hand in hand with this sort of trend of technology development, we have to accompany it with advanced material trends (Figure 5). We see on this pictorial chart at least how various materials have been introduced to allow us to move to either end of the scale, either to a higher allowable temperature or higher stress or a combination of the two. We have gone from stainless to titanium to cobalts and nickels at one end of the scale and composites at the other. These material trends are in fact making possible a good deal of our technology trends.

Accompanying the introduction of these new materials we must innovate, develop, and find new processes (Figure 6). I list here a few of the more advanced processes going into our gas turbines today. I would make one comment and try to keep it on an unclassified basis but in reference to the discussions earlier this morning about our competitors across the sea, it is our considered opinion that our friends have been able to develop many of these processes and that they are far from lacking in their ability to develop new materials, new processing, and particularly advanced design in their power plants. So I think -- I have to take a strong exception to anyone who says they are simply building on an old technology. That just isn't so.

Now, something interesting has developed over the years. We show in Figure 7 an index of effectiveness, if you will, for gas turbine engines. In this case we are plotting the power to weight ratio, divided by the SFC versus year of MQT. We note that we have shown our engines and competitive engines and they tend to fall pretty well on this straight line--the improvement line which is about ten per cent per year, with turbo jet and turbo fan engines being able to hold it to this trend pretty well through the years, as you can see. We can therefore expect to continue on it. Now, one odd thing about this is that when we look at turbo-shaft and turbo prop engines they don't fall on this trend (Figure 8). If we examine the trend as exhibited by two GE engines we see that the turbo-shafts have a lesser trend and this has fostered a general attitude that turbo shafts are sort of poor orphan childs. Now there is a good reason for this. The turbo shaft duty cycle, is different than the turbo fan or the turbo jet. It is just subjected to a much tougher job in the installations where it is used, and as a result, I think inherently one way or another we try to compensate for that by building into the turbo shaft a certain amount of margin, generally lower temperatures, and a certain amount of safety factor, if you will, to allow for the abuse that the turbo shaft has to take. And, this has continued up to nearly the present time.

However, there has been a considerable amount of interest recently expressed to us as well as on our part to change this trend, (Figure 9) to take advantage of the high temperatures and high cycle efficiencies that have been inherent in the turbo jet and turbo fan, and we feel that we have now developed a return to the original trend as shown by this increased slope line so that we predict, at least, based on some of our own advanced designs, that the turbo shaft of the 70's will tend to get back on the original line. We identify as our own personal goal - that box up in the corner - advanced technology which is getting very close to the original trend line. In other words we are going to try to have a renewal, if you will, in turbo shaft technology because we feel that there is still great potential there and perhaps that it has been lost somewhat in the conservatism of some of the turbo shaft designs.

In Figure 10 we find SFC improvement as going up the vertical scale versus horsepower to weight ratio. We want to show the general trend over the past few years. You see first that in 1960 we operated at moderate cycle pressure ratios and the horsepower to weight ratio actually would increase with pressure ratio up to a point and then we would begin to lose ground if we tried to achieve any further SFC improvement. The turbine inlet temperature and cycle pressure ratio trend go hand in hand. The two of them build together. You can see the potential of moving into this 1975 period where we feel that it is achievable to get 30 per cent SFC improvement and 50 per cent power to weight improvement.

Now, you may have noted previously that our goal in 1975 - 1980 period is a total index of effectiveness of 40 which would result from a horsepower to weight ratio of about 10 and an SFC of about .4. You will note here we have in Figure 11 the T64 and our four family engines which we are now developing. The reason for engine "D", the GE 12, being rather low on the horsepower to weight ratio scale is because with a rather small engine -- it gets difficult to compete with the larger engines on an absolute horsepower to weight ratio. The SFC trend is shown here clearly, and we see no reason why we can't get down to the level indicated there.

The question is how does this technology pay off. Well one example shown in Figure 12, is for a utility type helicopter, a UTTAS, if you will. We see that the payoff in the required vehicle gross weight to do the job could be as high as 30 per cent. That is a reduction in systems costs - I think that is one thing that we are talking about achieving in a VHLH.

We also try to show this on a typical heavy crane helicopter (Figure 13). These studies we have made on our own. You will note that the payload has increased about 10 per cent; however the radius can be increased greatly. The reason is the fact that the heavy lift is carrying such a large percentage of payload and not very much fuel. Also gains can be made in the fuel logistics area and they become more dramatic. This new technology gives load carrying potential of about 10 per cent over today's technology. This is not to imply that we can't build a helicopter with a much greater payload if you are willing to put a large enough engine in it.

These are a couple of "marketing" kind of charts (Figures 14 and 15) but I think that for those that aren't aware of our GE helicopter experience, here is a series of photos of the various T58 applications. I think we shouldn't minimize the fact that in each and every application we find out new things. We discover new problems and we work out those problems. They are all different and they all represent a certain amount of challenge and I think that experience in this VTOL field is a very strong factor in building the right engine. The T-64 engine family which is running from the original 2600 horsepower up to as high as 4800 horsepower now power these types of vehicles as shown: Compounds, STOL, turbo props, VTOL types, helicopters and even the hot cycle rotors shown on the lower left.

Now there has to be a certain practical input to this technology (Figure 16) as well and that is one of the reasons why I showed the pictures of all those vehicles because as we get into an operating environment we may find that we missed some critical aspects in the original design. We found, for example, that in the Army environment certainly erosion is a big factor and that we would not build another engine for the Army without erosion protection built-in. We found in our service with the Navy that corrosion is a big factor and we would design for that. We found a great deal of fuel contamination and we now design for that. Maintainability/survivability were discussed (Figure 17). I think survivability particularly in the area of vulnerability is a subject that is getting a great deal of attention right now. We are staffing certain organizations to work solely in that area. We are running a good deal of tests, firing at engines, examining--critical parts. Firesafety is one of the very strong factors that influence vulnerability. The duty cycle is quite important and you define the duty cycle very early in the design game. It has a strong impact on the total engine design, and finally growth, which we have classically and we will always build into the engine design.

Figure 18 shows, schematically, at least, the General Electric's concept that we will provide an integral separator for future turbo shaft engines. I think most of you are aware of what this is. It is basically a swirl-type vane which centrifically separates out small particles of dust, which are collected in collection scrolls and discharged overboard. We are trying to separate out very small dust, on the order of magnitude of the talcum powder that may be found in your wife's handbag; ten microns to 100 microns which can do damage and erode blades. We have to get it out. We feel that although there are many applications at the moment using separators of other designs such as the Donaldson-tube type---that the best overall design would be one that would be fully integrated with the engine for minimum power loss, having high cubic feet per minute per square foot, for front and rear drive, anti-iced and integrated with the engine front frame. I might say that we probably would be prepared to quote engine power with the inlet separator installed which would negate the reason for the 95 per cent requirement.

Survivability (Figure 19), we want to design for that and I think IR suppression becomes a factor now in engine design. Certainly we need fire safe low-pressure fuel system, low smoke, shock proof stack lines, self-contained lube and electrical systems and lube loss tolerance. What that implies is to have built into the engine the capability to run without oil for some short period of time.

Perhaps that ties-in with the requirement which implies that you must be able to fly with no engines operating to a safe area. I think we would tie in that you must be able to fly the engine with no oil to a safe landing area. We have been doing a lot of studies on grouping accessories on an engine in such a way as to get effective blockage, if you will, from small arms fire for critical parts by surrounding them with less critical parts (Figure 20).

Now here (Figure 21) are two representative duty cycles. This chart illustrates, perhaps, why turbo shafts, in general were on that reduced technology trend. The turbo shaft in a helicopter is exposed to that upper type duty cycle - it is a very good example. You can see the many transients, the high time at maximum power. That is just tough on an engine. And, when we design that duty cycle into our development runs, something has to give, either weight, stress levels, or life, or we backoff on temperature. The fixed wing cycle shown below is much less severe and we wish that everyone would recognize this and that a good deal of thought would go into how you are going to ultimately use the helicopter. We must define a realistic duty cycle. Recent inputs we have been getting in programs like AMSA define a set of wartime missions but use the vehicle on a peacetime basis and therefore we design the engine for the peacetime operation with the capability to perform the wartime mission but don't compromise the design of the engine for a continuous wartime type role.

Figure 22 shows the classic type of growth that has been designed into gas turbines and probably will continue. We have a potential for growth that's first developed from increasing the turbine inlet temperature, which is really a very minor change in hardware. And, finally, the usual thing, the addition of zero stage increasing the pressure ratio, and air flow. This trend shows examples of real cases where the engine can grow one hundred per cent in a ten-year life.

A FAMILY OF ADVANCED TURBOSHAFT ENGINES

GENERAL ELECTRIC

AIRCRAFT ENGINE GROUP

LYNN, MASSACHUSETTS/CINCINNATI, OHIO

FIGURE 1

MARKET NEEDS - Size ranges for 1970-80

SEA LEVEL HP	SYSTEM
■ 400 - 800	Light observation helicopter Utility observation aircraft
■ 1000-2500	Troop carrying helicopter ASW helicopter ASW attack helicopter
■ 4000-6000	Light close support aircraft Medium assault helicopter Rescue Helicopter
■ 6000-10000	Light STOL transport Troop carrying helicopter
■ 8000 and up	Heavy crane helicopter

FIGURE 2

THE SPECTRUM OF ADVANCED SHAFT ENGINES

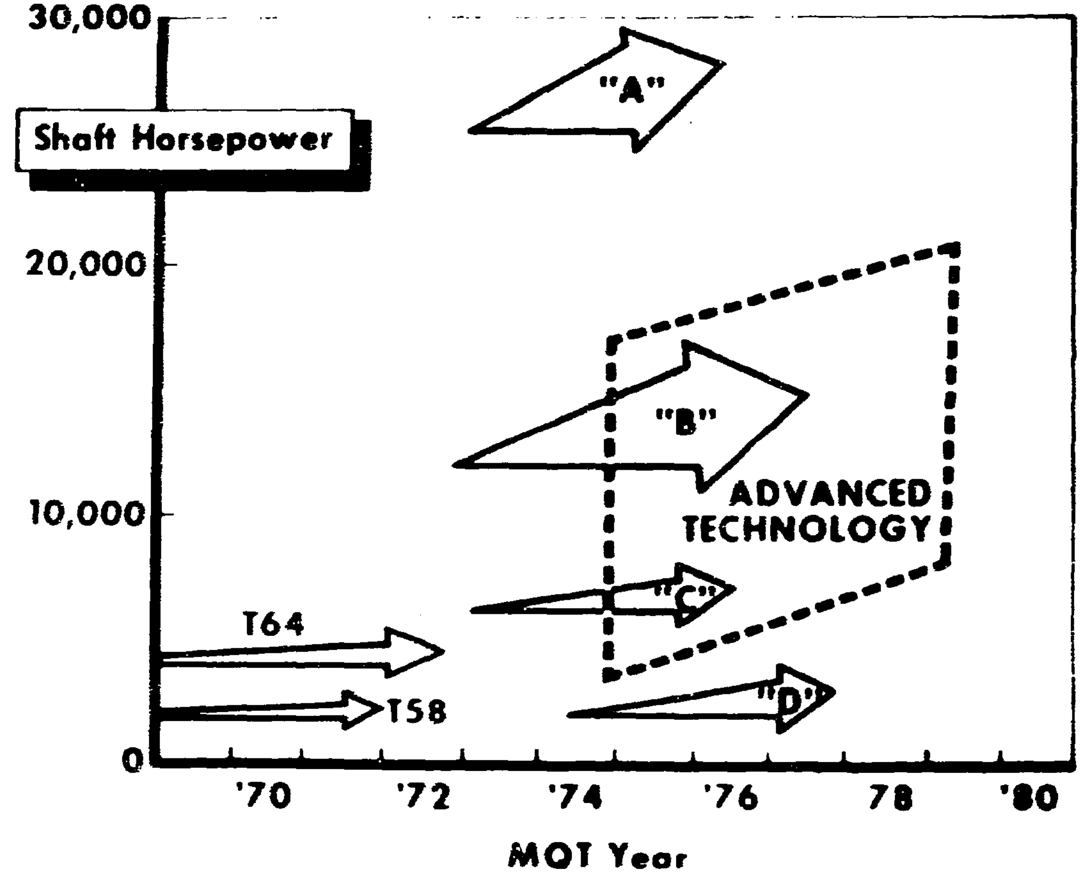


FIGURE 3

TEMPERATURE & PRESSURE RATIO TRENDS

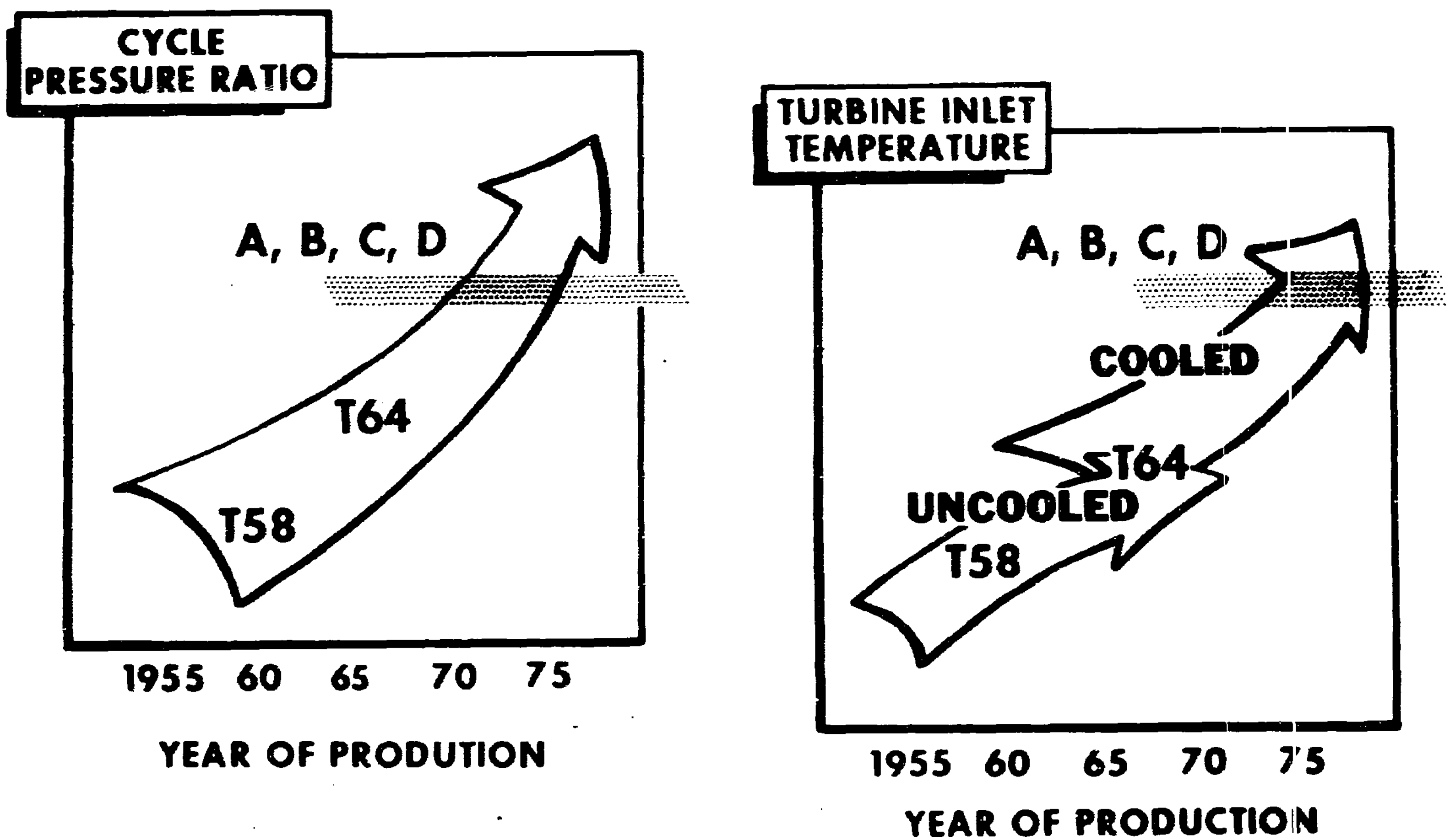


FIGURE 4

MATERIAL TRENDS

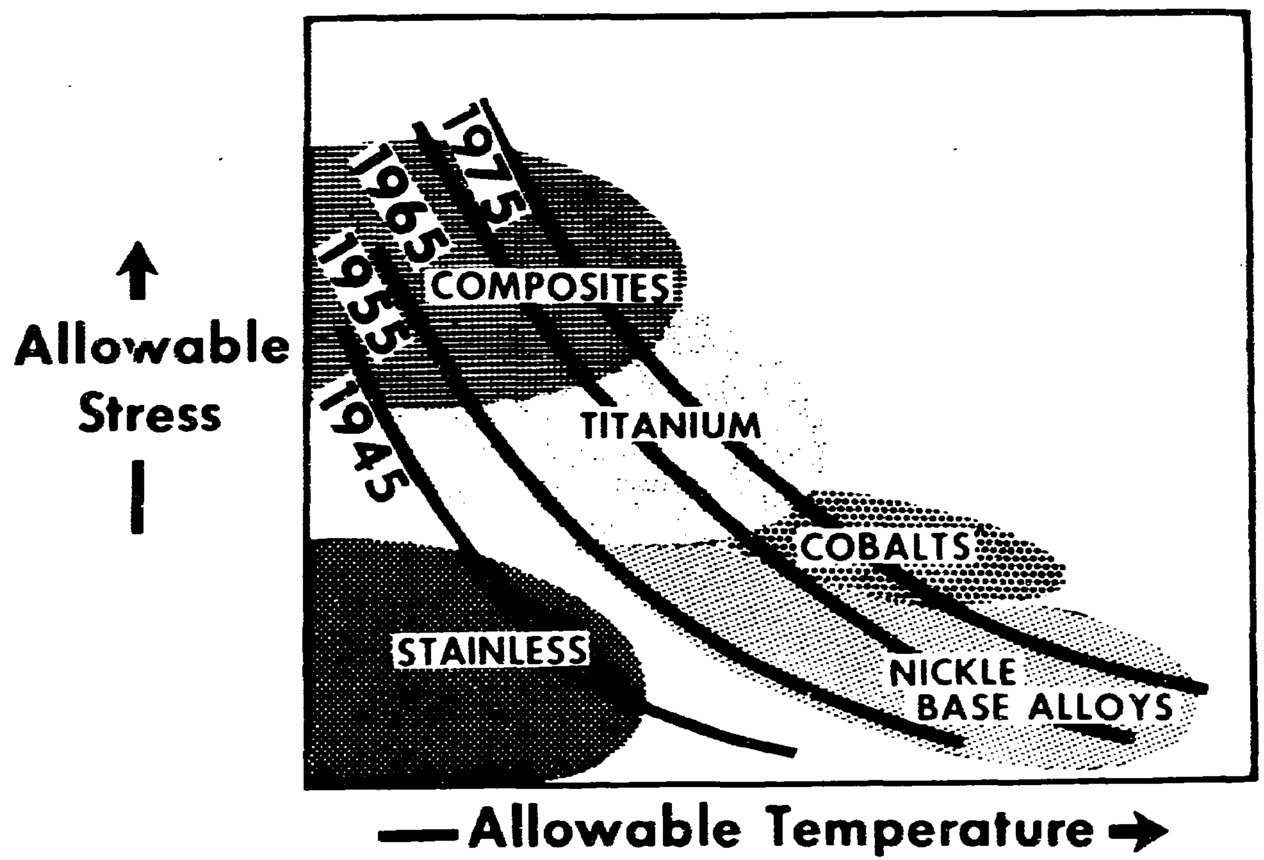


FIGURE 5

PROCESS PROGRESS

- Diffusion Bonding
- Impact Welding
- Roll Forming
- Electro Stream/Laser Drilling
- Single Crystal Casting
- Composites
- Coatings

FIGURE 6

TREND OF ENGINE IMPROVEMENT WITH TIME

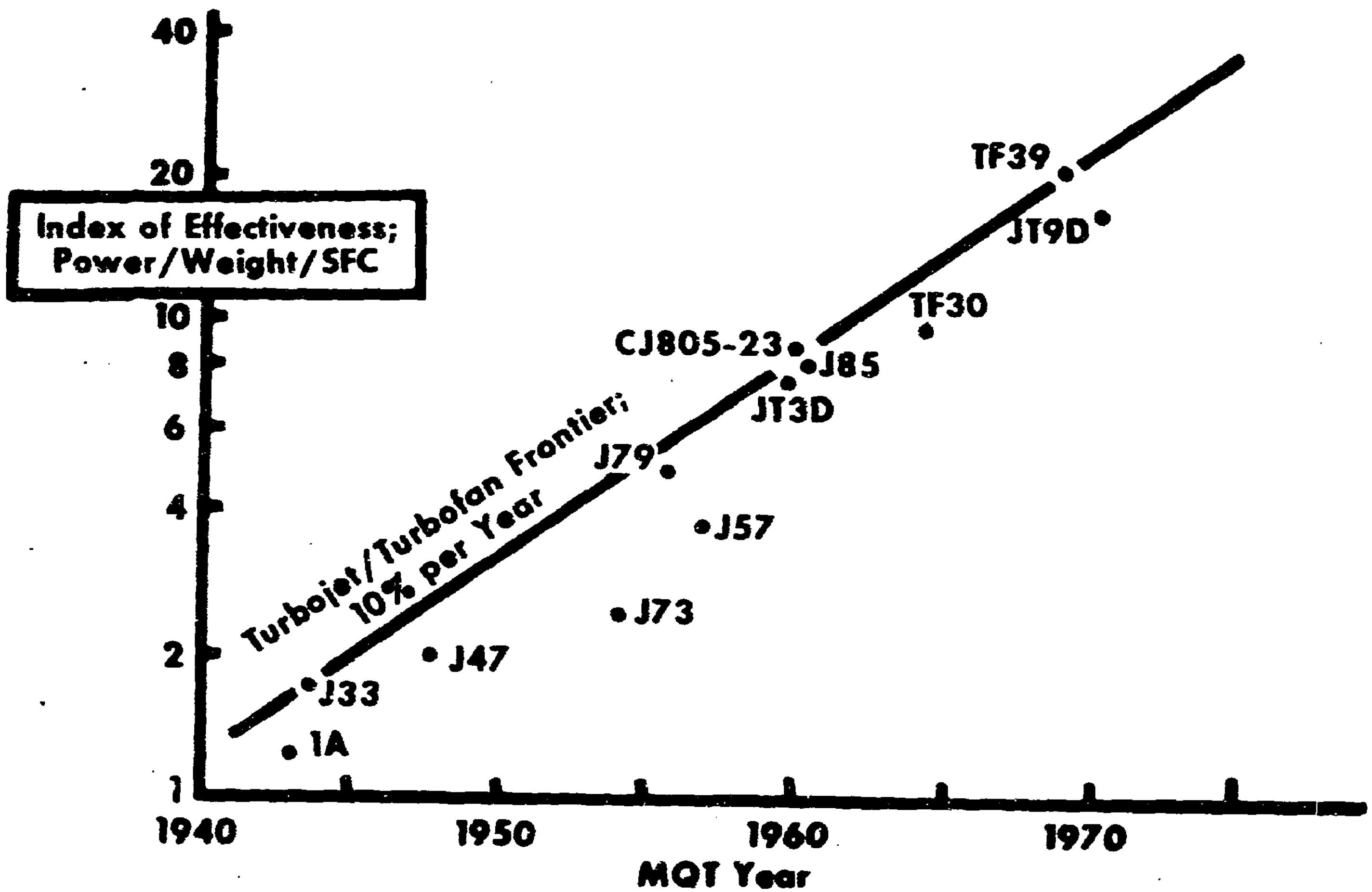


FIGURE 7

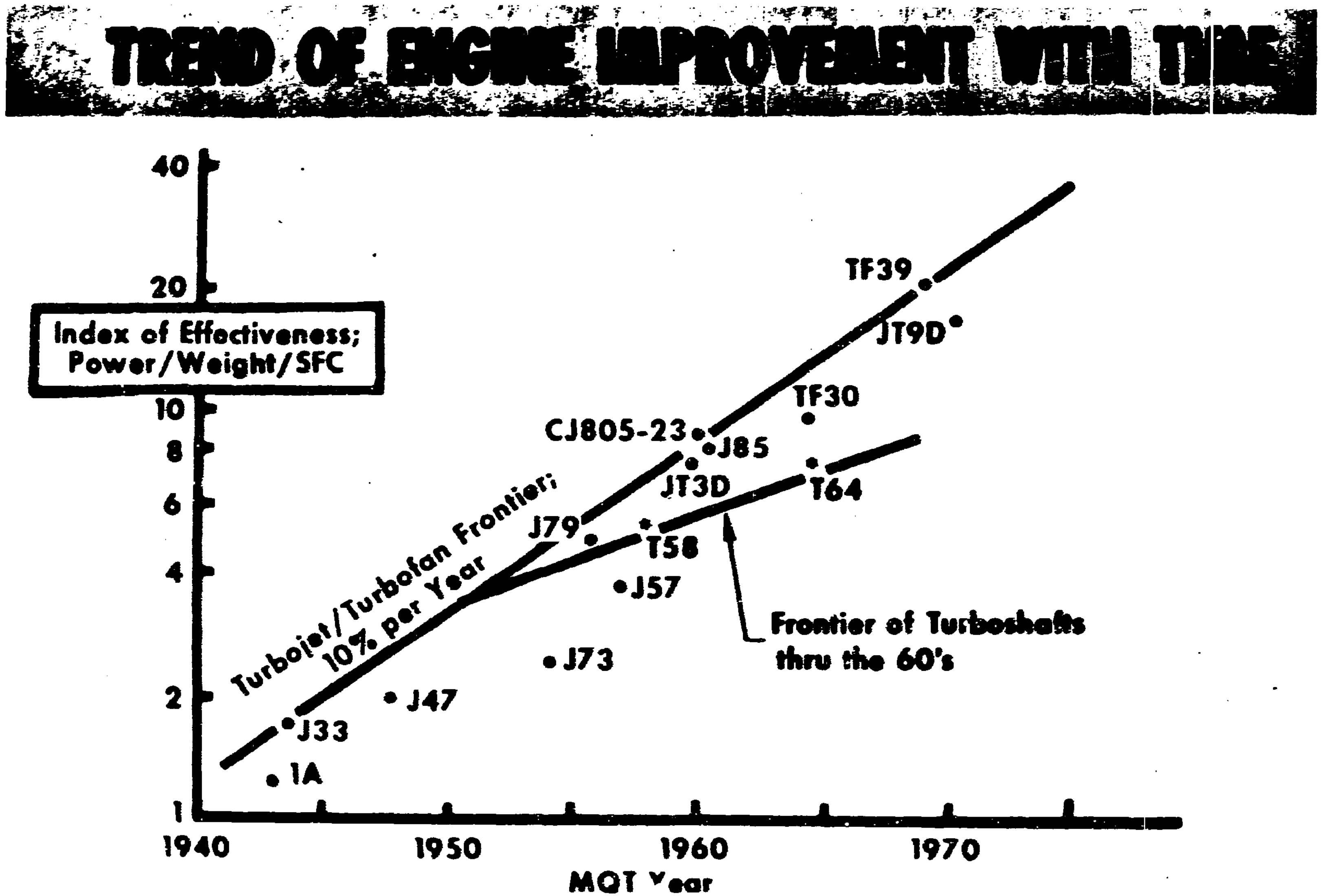

TREND OF ENGINE IMPROVEMENT WITH TIME
Index of Effectiveness;
Power/Weight/SFC
40
20
10
8
6
4
2
1
1940
1950
1960
1970
MQT Year
Turbojet/Turbofan Frontier;
10% per Year
Frontier of Turboshafts
thru the 60's
TF39
JT9D
TF30
CJ805-23
J85
JT3D
T64
J79
T58
J57
J73
J47
J33
1A

FIGURE 8

TREND OF ENGINE IMPROVEMENT WITH TIME

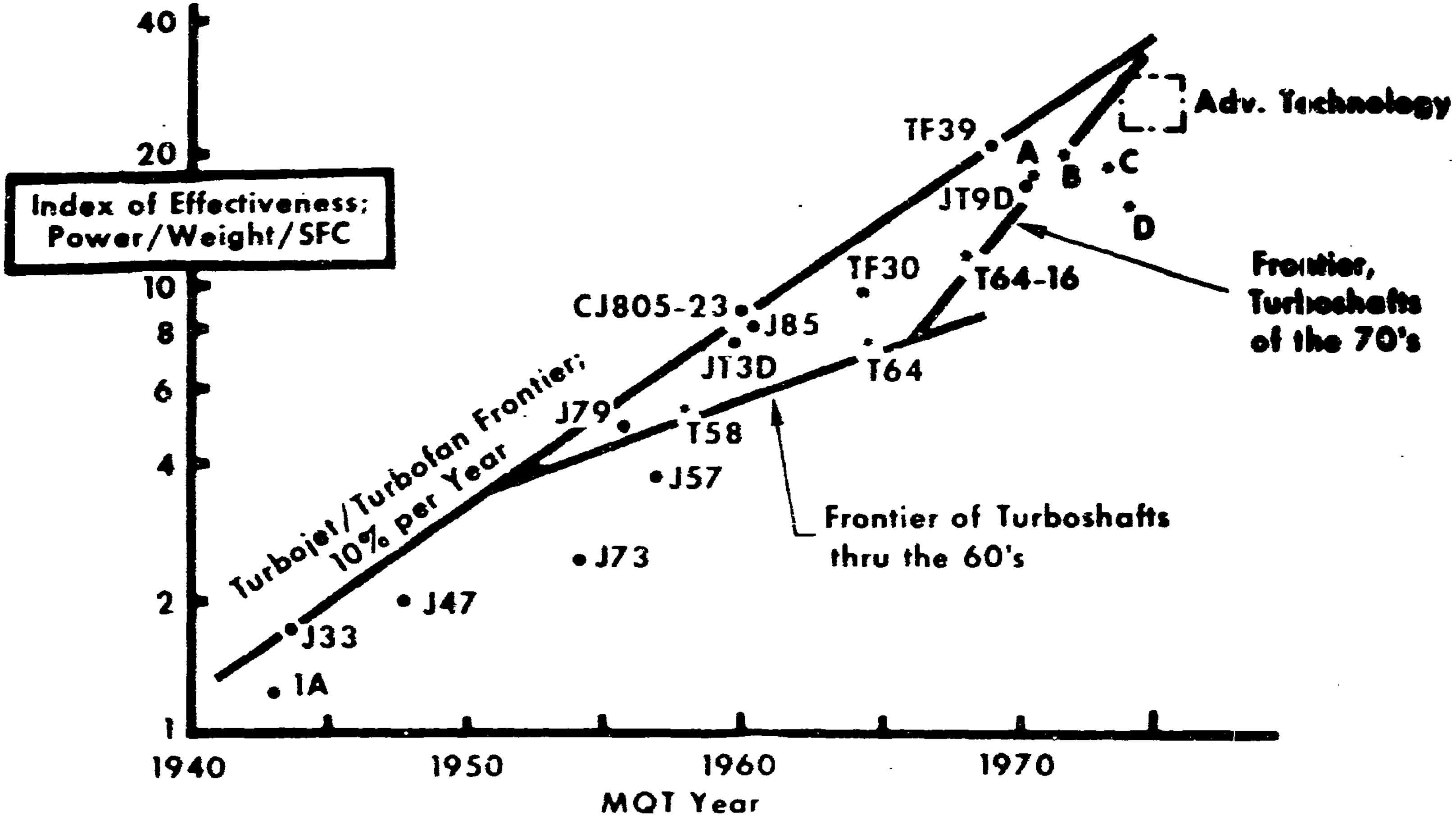


FIGURE 9

INFLUENCE OF CYCLE PARAMETERS ON SHAFT ENGINE PERFORMANCE

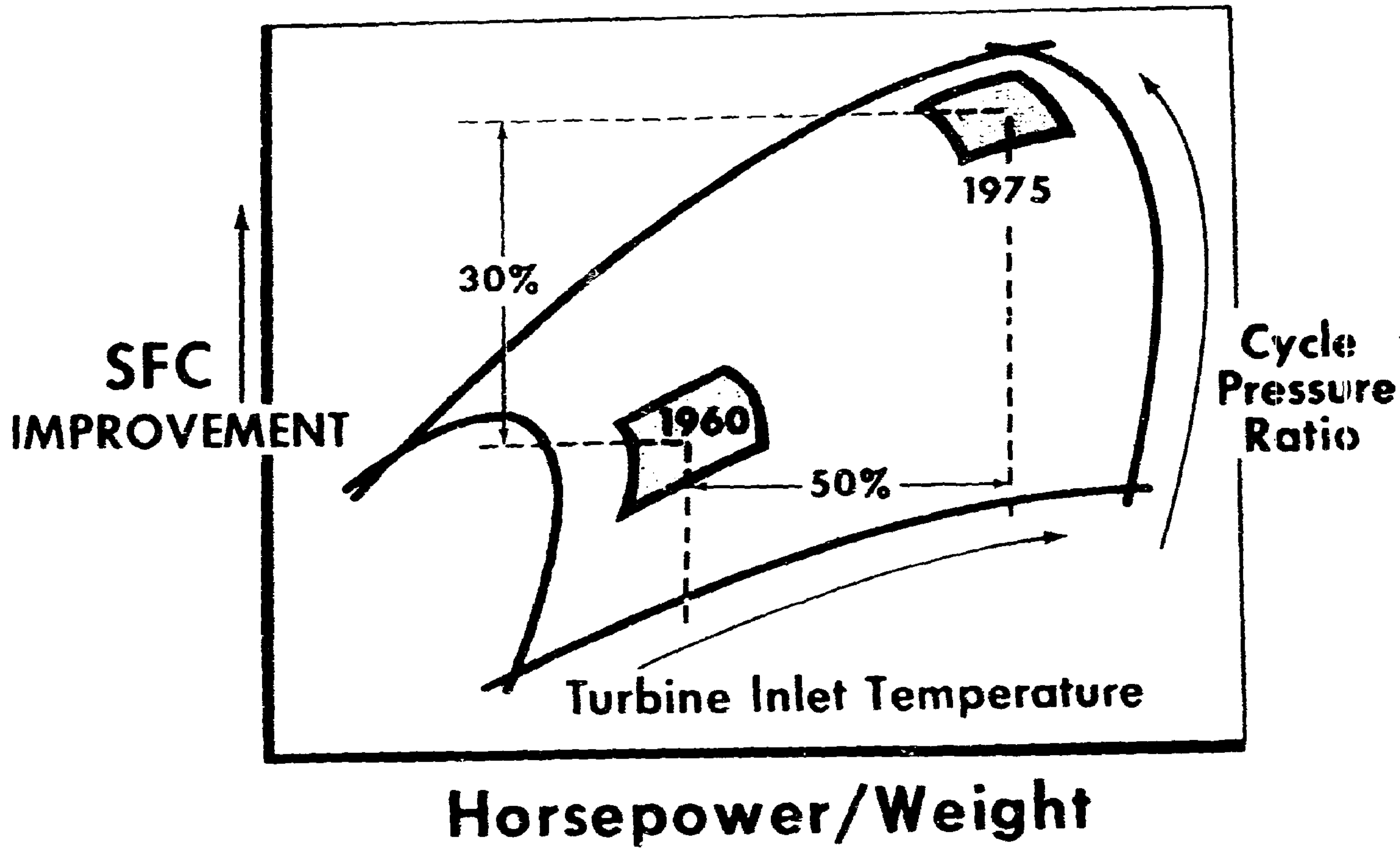


FIGURE 10

HORSEPOWER/WEIGHT & FUEL CONSUMPTION TRENDS

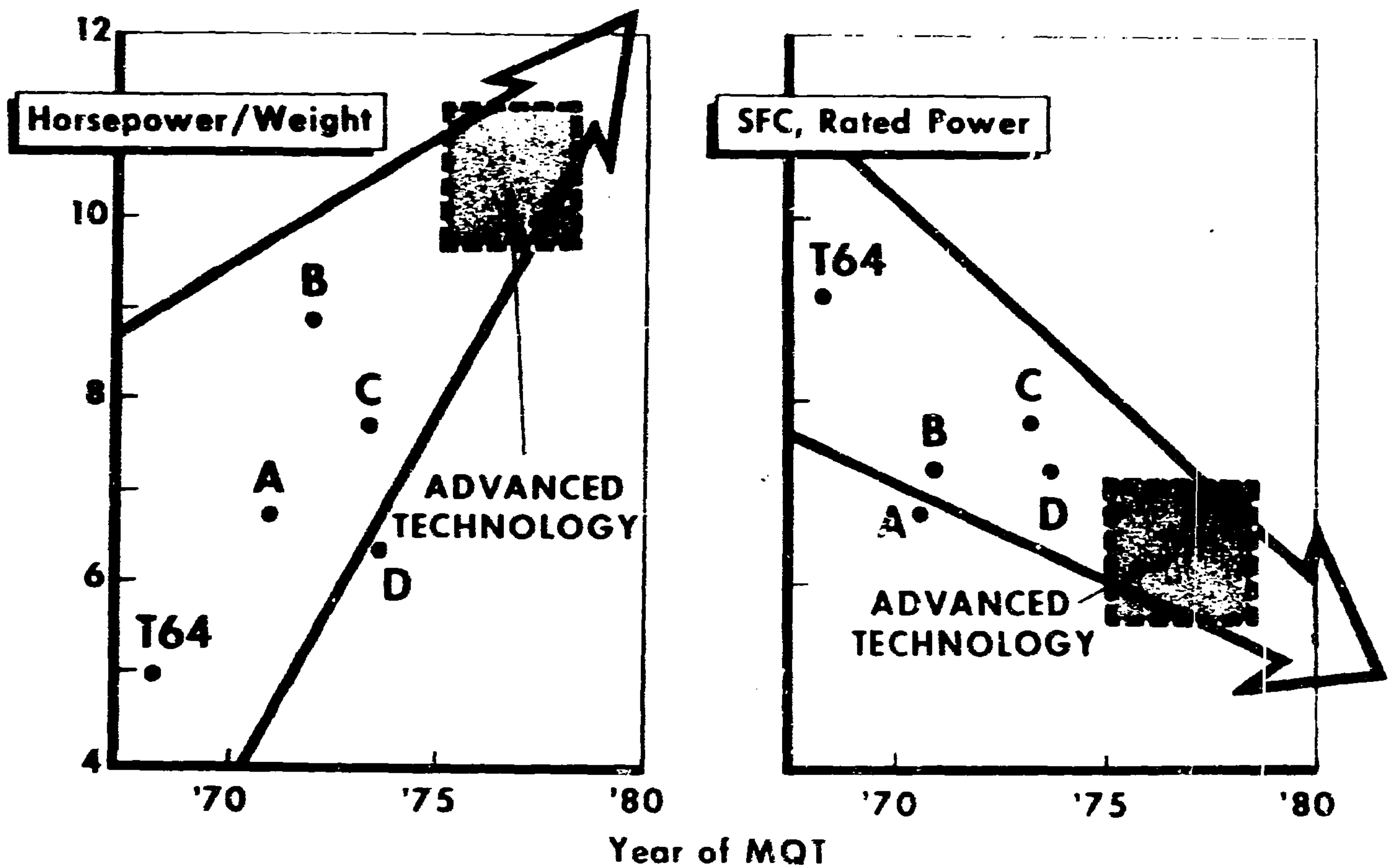


FIGURE 11

TECHNOLOGY PAYOFF; UTILITY HELICOPTER

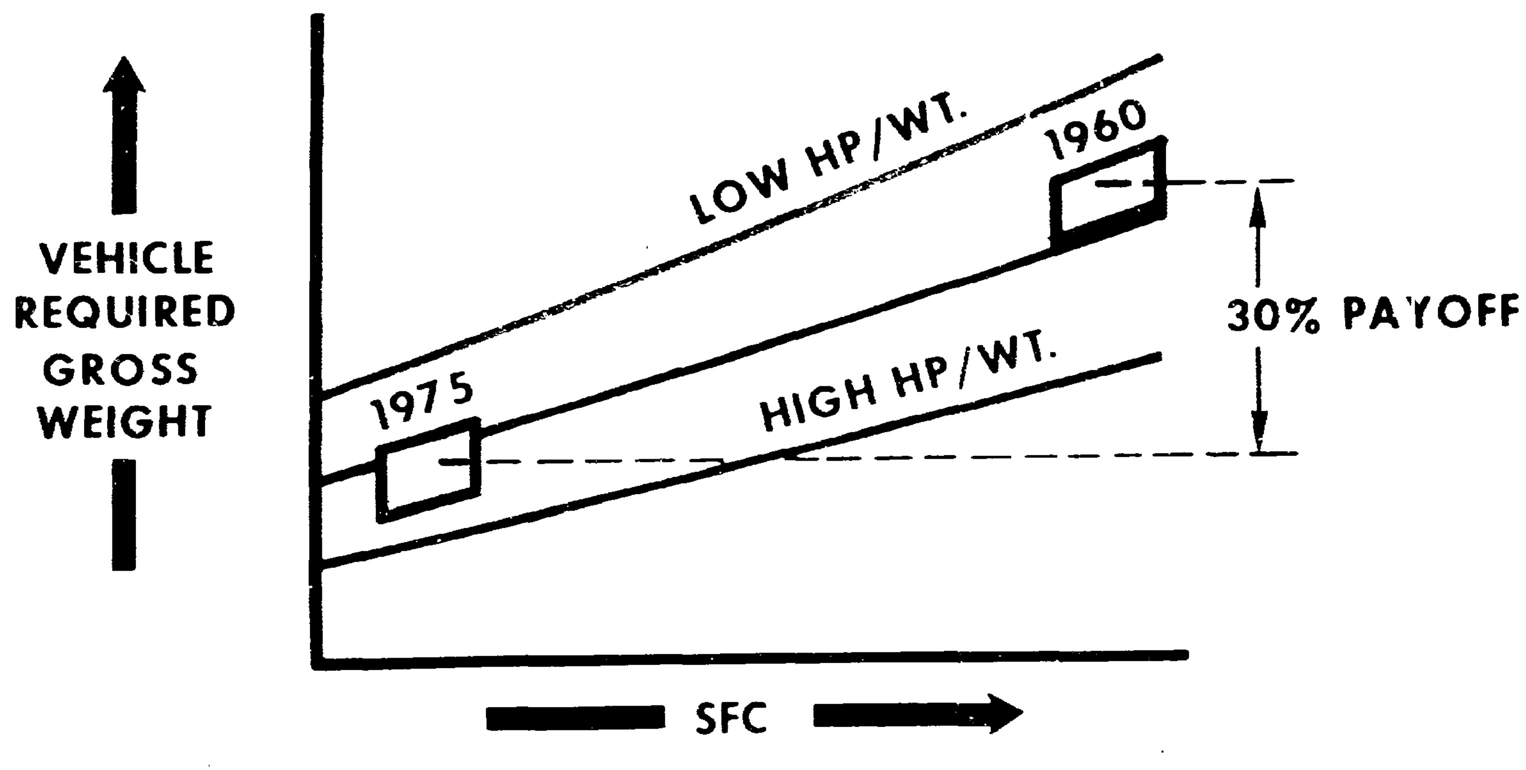


FIGURE 12

TECHNOLOGY PAYOFF, HEAVY CRANE HELICOPTER

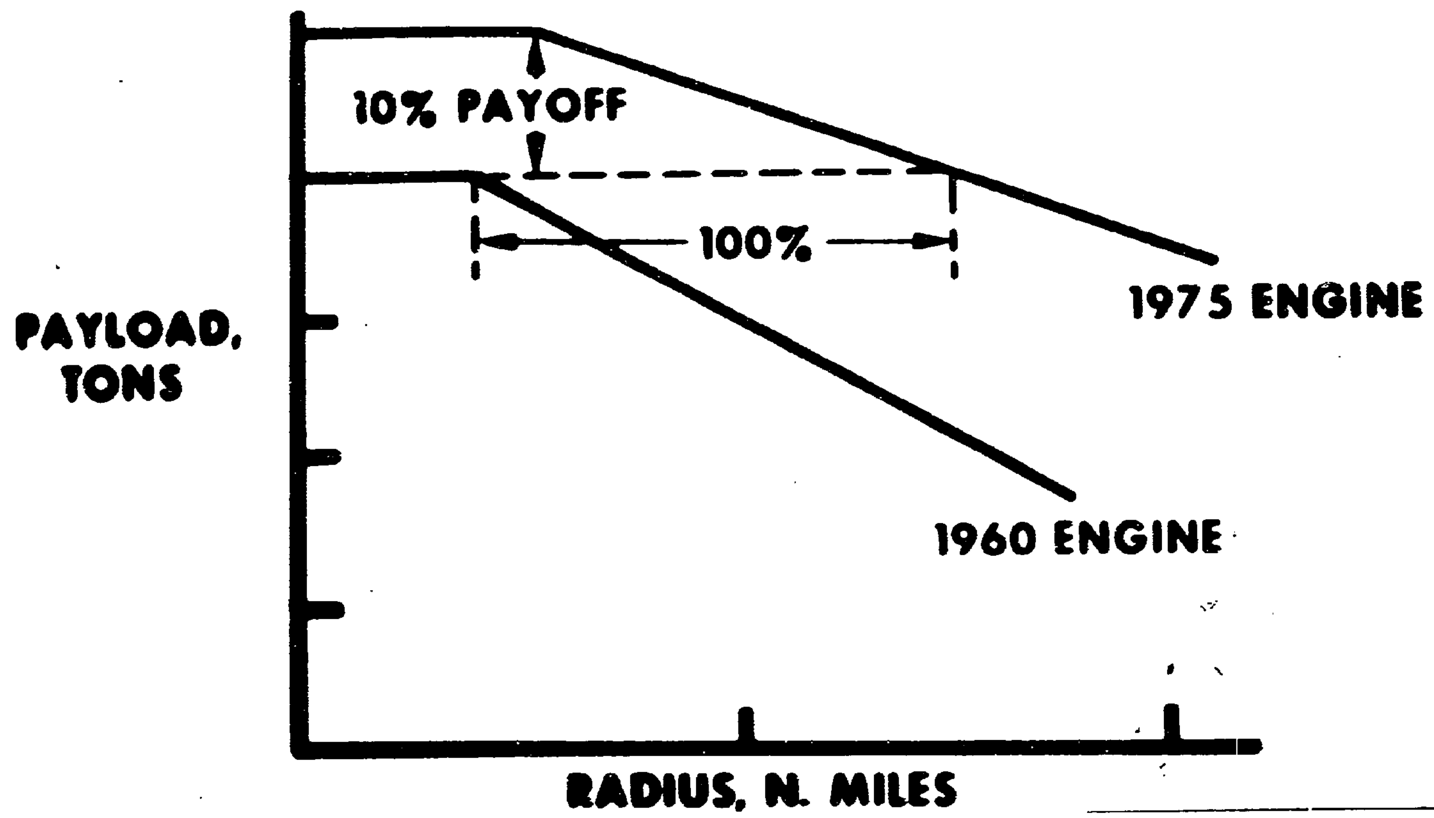


FIGURE 13

PAST EXPERIENCE...
DESIGN AND INSTALLATION

T58
FAMILY

FIGURE 14

PAST EXPERIENCE . . .

DESIGN AND INSTALLATION

T64 FAMILY

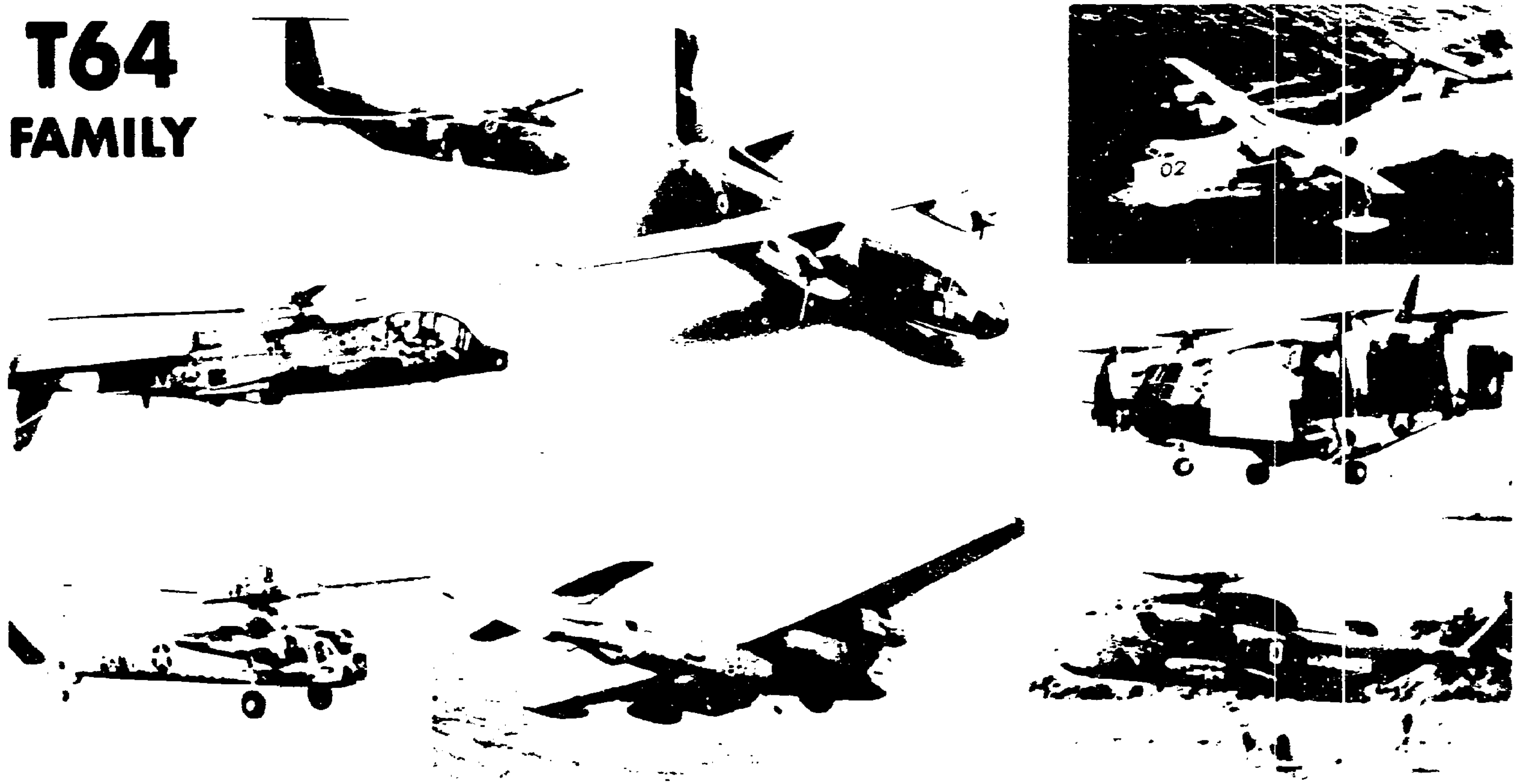


FIGURE 15

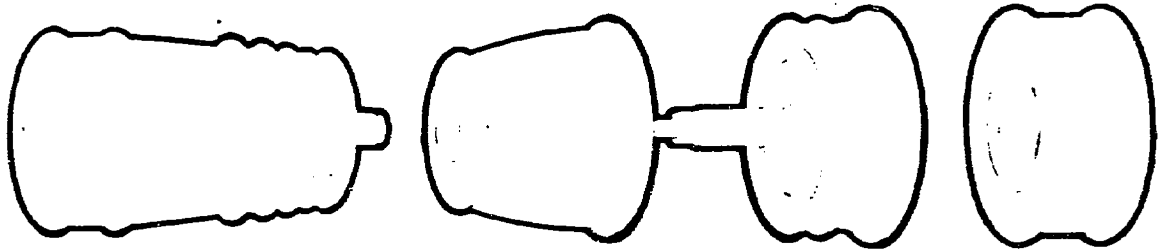
THE PRACTICAL INPUT TO TECHNOLOGY UTILIZED

- Environmental Factors
  - Erosion
  - Corrosion
  - Contamination
- Maintainability
- Survivability (Reliability, Vulnerability)
- Duty Cycle
- Growth

FIGURE 16

MAINTAINABILITY IN THE FIELD

- Modular construction



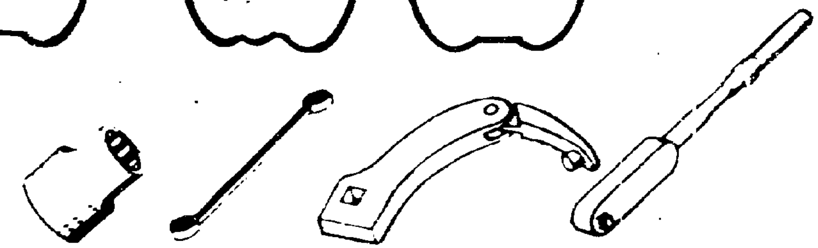
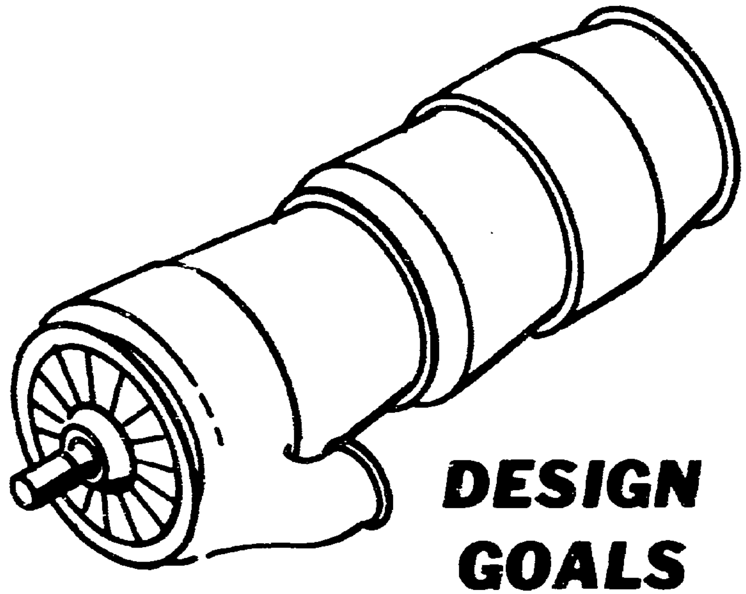
- Standard tools 
- Replaceable blades, buckets, vanes, partitions
- Diagnostics

FIGURE 17

SCHEMATIC OF AIR PARTICLE SEPARATOR



DESIGN GOALS

- ***Minimum Power Loss***
- ***High CFM/Sq. Foot***
- ***Front or Rear Drive***
- ***Anti-Ice***
- ***Integrated with Engine Frame***

FIGURE 18

SURVIVABILITY

- IR suppression
- Fire safe - low pressure/suction fuel system
- Low smoke
- Short, grouped, stacked lines
- Self-contained lube and electricals
- Lube loss tolerant

FIGURE 19

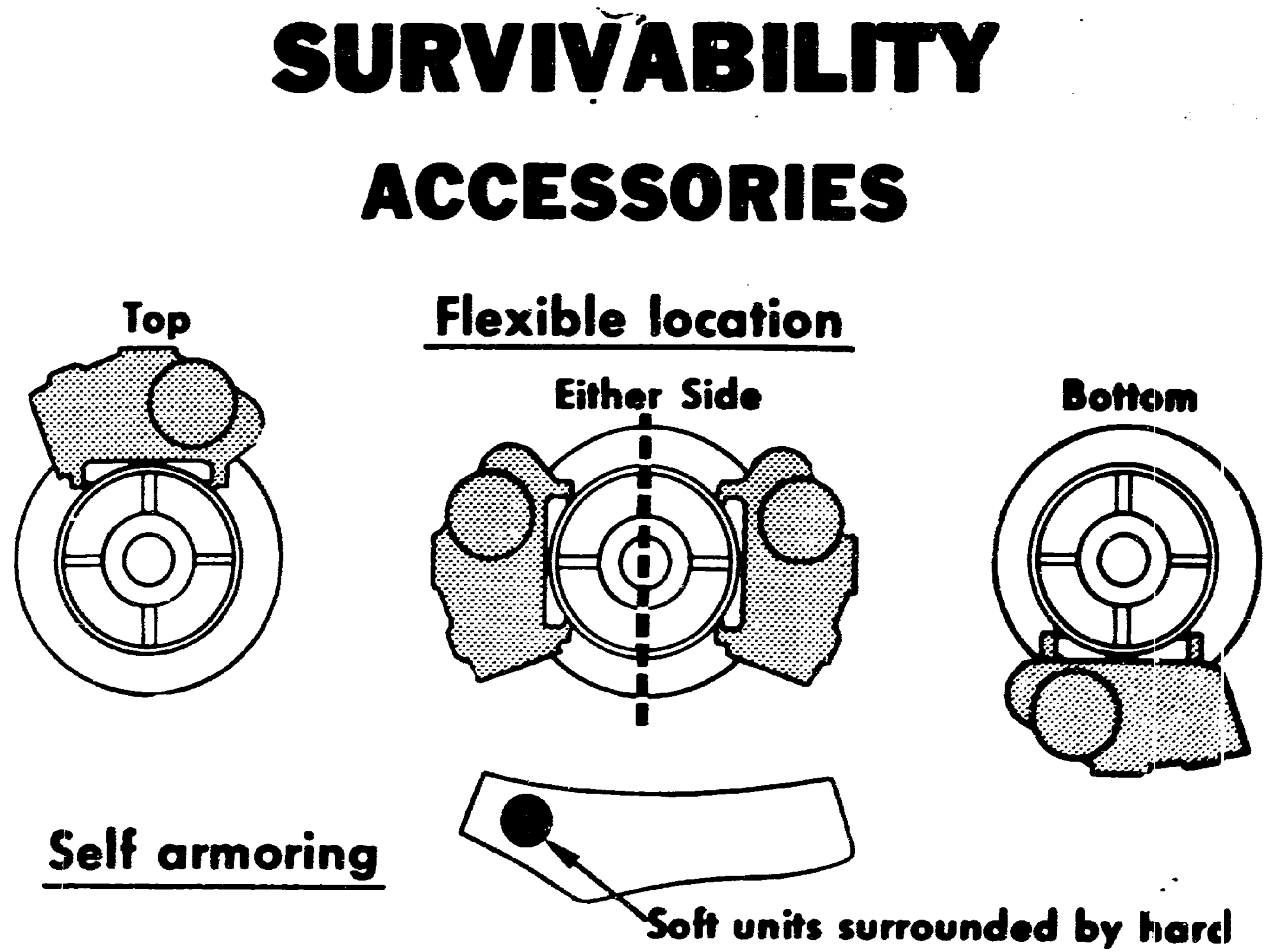

SURVIVABILITY
ACCESSORIES
Top
Flexible location
Either Side
Bottom
Self armoring
Soft units surrounded by hard

FIGURE 20

TYPICAL DUTY CYCLES, SHAFT ENGINES

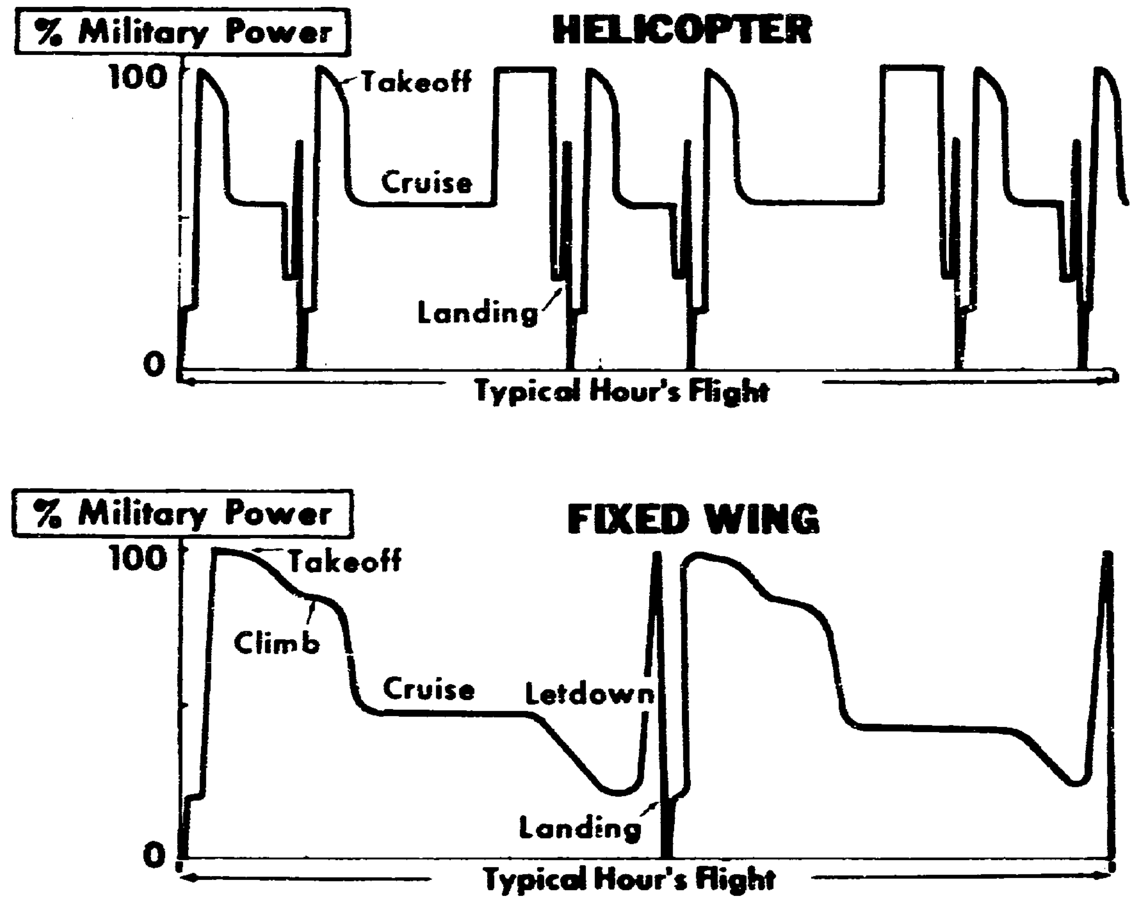


FIGURE 21

GROWTH

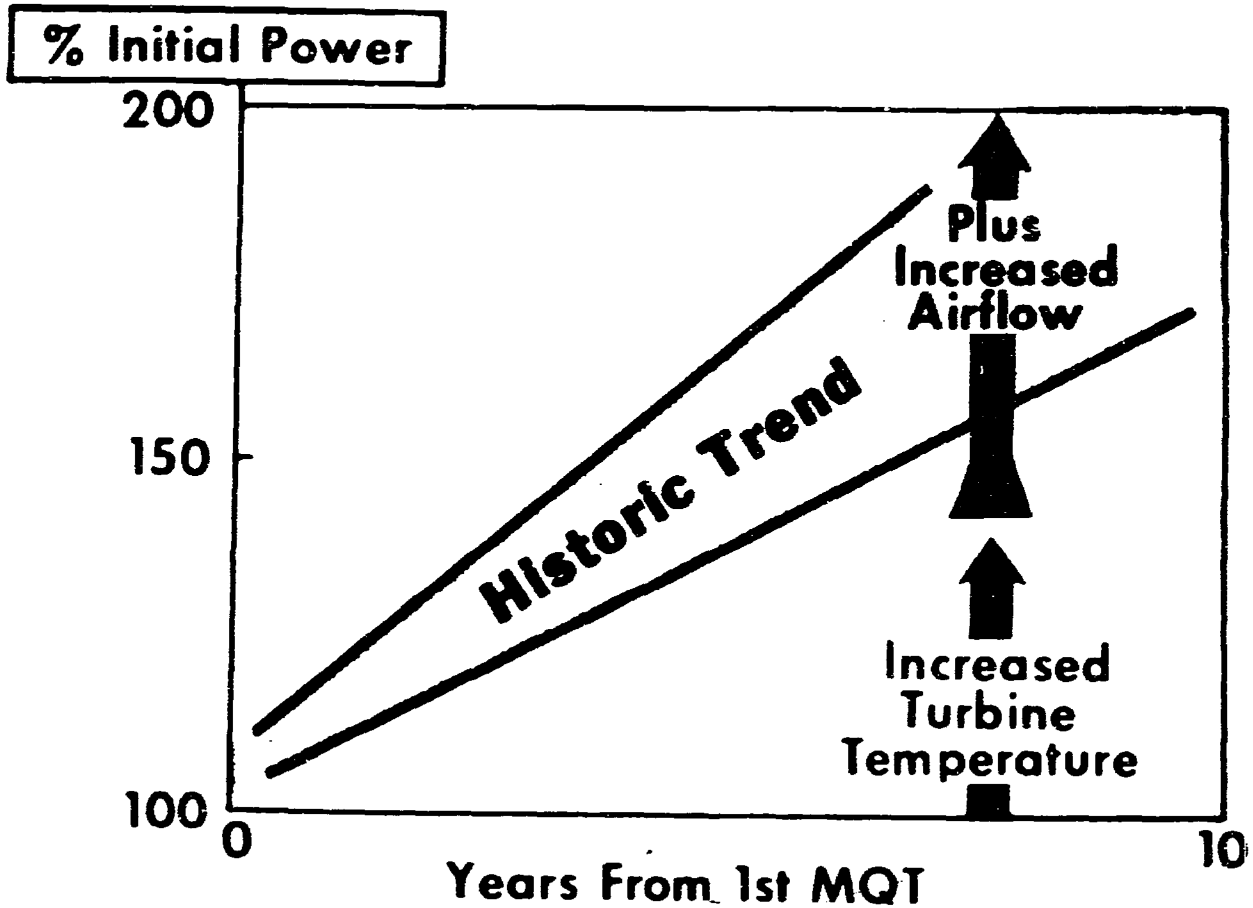


FIGURE 22

APPENDIX I
SHAFT DRIVEN VHLH TECHNOLOGY - 1990

Presentation by Lewis Knapp - Sikorsky

Some gross weights for a 50-ton payload VHLH are going to be shown which are somewhat different from those Dr. McCormick showed, since I've been looking forward to 1990 technology. If the VHL is in fact a helicopter, and if it is a shaft-driven helicopter it will in all likelihood have two rotors. If the two rotors are single lifting rotor and tail rotor the VHLH could very well be configured as the model which some of you saw at the AHS last week. (Figure 1.) If it were a coaxial rotor system, it might look somewhat like this. (Figure 2.) A twin rotor system, we have shown here, actually, a way-out idea for a twin rotor system which has our structural dynamicists pretty ill of course. (Figure 3.) This particular design was based on a study we did, whereby we said, "If the 'going in' cost for something this size is too terribly high, why not take the dynamics from a smaller size helicopter and use them in multiples on a single aircraft?" We got conclusions which were not as good from a weight standpoint as a cleanly designed 50-ton payload tandem, but if the total buy is small, it might be a way to go. Of course, if the going in cost is too great to be viable at all, the job can be done with two rotor systems like this. (Figure 4.) There is a possibility that the total number of very large payloads, or the total number of times when these very large payloads have to be carried are so few that a pair of, say, 30-ton aircraft acting in multilift, can carry a 50-ton payload. Whether, in fact, a 50-ton machine is needed is going to depend upon the number of times that it needs to be used and the importance of that mission.

Using some of the preliminary designs that we have put together from time to time, I have compared a single rotor aircraft (using what we have assumed to be 1990 technology) with a coaxial rotor system, and I would like to give you some of the results of that comparison. In comparing the gross weights of a single rotor with a coaxial rotor system we came out with some interesting numbers. The single rotor aircraft, for a 50-ton payload, 1990 technology, grossed out 207,000 pounds. The coax was lighter, 192,000 pounds. The weight empty for the coax system, 76,000 pounds. The fuel for the single rotor, 15,400 pounds. Believe me, achieving this fuel weight is really going to depend on the engine technology Les Veno described because the fuel requirements for this 50-ton mission are not much greater than the fuel requirements for a 27- to 30-ton machine using 1969 technology. So, you are going to have to achieve that 0.40 SFC.

The fuel for the coax came out about 1,000 pounds less, or about 14,400 pounds. We also assumed that a disc loading of 10 was a good maximum number to use. This has been kicked around an awful lot. There is not yet in the industry a clear statement as to what is the largest desirable downwash velocity and therefore what maximum disc loading do you want to tolerate. So for this single rotor we assumed the disc loading is 10, and for the coaxial system we assumed the disc loading is eight and a half. The life cycle cost analysis showed lower costs at DL = 8.5 than at DL = 10 for a given type. The single rotor came out with 160 feet diameter and a coax of 154 feet in diameter. The installed power on a single rotor would be about 34,300 horsepower. On the coax, it would be about 27,000 horsepower. Perhaps, it is our ignorance that makes us this optimistic about the coaxial system's configuration, so I am merely reporting to you what our studies have shown, and I am not commenting on the risk of that type helicopter configuration in this size.

I mentioned that we assumed 1990 technology. Everything that we felt might be developed in the next 21 years to reduce the weight empty fraction and increase the payload fraction was incorporated in our studies. For example, we assumed 9,000 PSI hydraulic systems. We assumed redundant electrical flight control systems. These are weight savers and by the 1990 time period, the reliability of that kind of flight control system should be well established. The impact of technology on payload fraction is shown here (Figure 5) for a 20-ton payload aircraft. If that 20-ton payload aircraft used the same technology that was put into the CH-53 a few years back the aircraft would gross about 130,000 pounds. If we used today's technology, or technology which could be put into an aircraft which could be started today, that same aircraft for a 20-ton payload would gross about 112,000 pounds. By 1990 using the most optimistic improvements in technology that we could, the aircraft would gross around 103,000 pounds. So you see, technology has a very important effect on the gross weight for a given payload, or on the payload fraction.

Now let us talk a bit about size. We said that present-day configurations of aircraft can be extrapolated to 50-ton payload sizes by advances in component technology, and that advances in component technology are necessary to hold the line on component weight fractions. Examples of this might be in rotor blade systems. We at Sikorsky have used extruded aluminum alloy blades in our helicopters to date. We have made some studies for a 27-ton aircraft and this particular one happens to have 4 blades. We have made some comparisons of various materials for blade spars (Figure 6). The titanium alloy blade, as you can see, weighs less than 70 percent of the weight of the aluminum alloy blade. Studies of composite blades showed even further weight savings. But the fact is that on these very large machines the composite blade may be too light. On a free flapping rotor system - over at the other side of that chart (Figure 6) you see the coning angles at the design gross weight. The composite blade that we designed would cone at about 11 degrees. This means that it would be a high Mach Number blade, rather gust sensitive, and could not take a high load factor. Of course, there are things that could be done to improve a composite blade, like increasing the centrifugal relief by tip weights and things of this sort, but as you do this you begin to trade off your weight saving and you will find that an ideal composite blade with a relatively low coning angle achieved by addition of tip weights would in fact not be much lighter than titanium blades on these very large machines. Here is an approximate blade size on a 50-ton payload aircraft, assuming the gross weight is 207,000 pounds, as I mentioned, and assuming a 6-bladed rotor system. (Figure 7.) That blade would be 92 feet from rotor center on a single rotor aircraft, and would have a blade chord of close to 6 feet. This gives you some feel for the size of the blades we are talking about. In these very large sizes, we have a potential low frequency dynamic problem. The rotor that I am speaking of would rotate at around 89 rpm and you want your n per rev to stay far enough above seven cycles per second so that the crew was comfortable. This could mean that the rotor I am describing would be an eight-bladed rotor, not a six-bladed rotor as shown here. Therefore, for a given total blade area the blade chord would be down somewhat from what we have there. For very large blades, some work has to be done on titanium spar construction techniques. There are several possibilities open to us here. (Figure 8.) The ideal shape for a blade spar is a closed D section. One way that such a spar might be made is to extrude it to length, as shown here. Extrude it as a hollow billet. There is no capacity to do this in these lengths today, but the Cameron people in Texas are capable of making the blade forging - or extrusion. (It is actually a combination of forging and extrusion techniques.) They are making sections 40 feet long now, and with new capital equipment they see no reason why they can't go to much larger sizes. Such a blade might be automatically machined to a round tube with internal taper and tapered thickness as shown here. We have just completed having a series of titanium blade spars 20 feet long machined full length in a gun barrel factory, with thicknesses of 3/8 of an inch at one end and .080 at the other end with good control of the thickness. That kind of process could be available on a production basis, giving you a circular tapered titanium spar.

Alternatively, to get to that same point, a short length of extruded forging from someone like Cameron could be elongated by any of several hot flowing processes. This development in Ti could be brought out to the full length spar that we are after. As a next step, the round tapered spar can be hot formed in glass rock dies, or other dies to the final spar shape, and then built up conventionally with a non-structural trailing edge and a non-structural leading edge, possibly with tip weights and the anti-icing and so on. This spar forming process has been used at Sikorsky on 20 feet long spars.

Another potential method of manufacturing a blade which ought to be examined is the multi-piece Ti design where the blade is fabricated in two halves and joined, perhaps by roll diffusion bonding, perhaps by E.B. welding, again ending up with the single spar and then adding the airfoil shape. For 1990 technology I think that, in the 90 odd feet diameter, these techniques can be developed; so I see the capabilities of having rotor blades for a rotor of the size that I mentioned. As to rotor head, some work has been done over the years on new types of bearings and a rotor head might be configured like this. (Figure 9.) Once, again, because of the size, it would probably not be made as a single forging but separate independent billets might be diffusion bonded to each other and then the thing finally shaped. Or, North American Rockwell is working with diffusion bonding of rotor head plates. They have done a couple samples for CH-53 to date. There are technologies available that would give us rotor heads of the sizes that we are after. Shown on the rotor head is a spherical elastomeric bearing. This kind of bearing can serve as a combination flapping, lag, feathering hinge for the blade. There has been enough work done in the industry so that we all feel together that it is a viable technique for providing blade bearings and will replace the lubricated blade bearing. This is a small scale test that we did on the thing back in 1962 and 1963 (Figure 10). Since then more work has been done in the industry by Lord; Boeing-Vertol has done a very much larger size than this with considerable success and we think we have a good solution for the rotor head bearings.

So much for the rotor system. The other unique thing about the shaft-driven helicopter in very large size is its transmission system. We have talked a bit about the square cube law, but what is going on to date in the industry, is to prove that it is a law that can be violated-or maybe it isn't a law at all and technology improvements are needed in order to overcome it. If the aircraft I just described had a single rotor, its transmission system would have an 89 rpm output, and the final stage of the transmission would have to transmit 1,700,000 feet pounds of torque. Now this is 12 times the torque that goes through the last stage of the CH-53 -- that spells big gears -- if that were done with a bull gear driven by, say, 8 pinions -- the bull gear would be about 8 feet in diameter and the face width would be about a foot and a half, for example. At the present time in the industry, we can internally grind ring gears only to 48 inches. Equipment, larger than that is not presently available but some new equipment is being made. At the present time in the industry we can face harden gears only up to 36 inches in diameter if we want to use the carbonizing processing. So this would require new larger equipment, but the processes are available. In a very large transmission new fabrication techniques would have to be made available to meet the required weight fraction. We mentioned the square cube law; the torque does, in fact, go up as the three-halves power of the lift but historically propulsion weights have gone up as the three-quarters power of the torque. Therefore, the weight in the propulsion system will vary as lift to the 1.18 power if you don't make technological improvements. So what can we do in transmission systems to make them better than they are today? In transmission configurations, the roller gear drive, if it can be made to work, provides a very large reduction ratio transmission system. (Figure 11.) It promises to save, for a given reduction, 5 to 10 percent of the weight in present-day planetary systems and it does this because there are more free floating gears in the system and fewer overall bearings. It is not at the point where it can be put into production; R&D work is being done to find out whether this in fact is a lower weight transmission concept than those that we presently use.

In transmissions as in other helicopter components, if we can reduce the total number of mechanical joints and replace them by other types of joints we achieve greater efficiency of material usage and reduced weight. In transmission systems by employing diffusion bonding so that large gears are built up, (probably even diffusion bondings of dissimilar materials), instead of being made of a single forging. You can put the material where you need it, and you can assemble the gear by one of these non-mechanical joining techniques, ending up with a light stiff gear system. E.B. welding has great promise for use in transmission systems. For gearbox housings, to date, we have used magnesium castings. They are relatively light and they are relatively stiff, but this is not really an efficient use of material, and as we go to larger and larger sizes, forgings will be welded together for the main structural members, to provide stiffness where the materials are needed, and a thin shell will be provided to keep the lubricants in. New light weight materials will be used. I think there is application for beryllium compounds in main gearboxes, for planetary plates. Again, work has to be done and things have to be thought out before we are ready to say this can go into production. For the far out time period, transmission R&D should also include examination of lubricants and lubrication systems for greater lubricity, for reduction of the potential of catastrophic failure after loss of lubricants - to provide transmission systems which will last a half hour or an hour after the lubrication is gone. Perhaps this means redundancy in lubrication systems, but we need to look at the lubricants, the lubrication systems and the cooling systems and in all of these areas I think that we can make improvements that will reduce weight and at the same time improve reliability of the system. We want to take a good strong look at gearing tooth loadings, new materials, perhaps new tooth shapes. We would like to try to improve the efficiency of the transmission systems. The main gear box loses into heat about two and a half percent of the power transmitted from the engine. It is not a large total percentage but we would like to be able to reduce it. And, of course, in bearings, who knows but what in the future non-liquid lubricated bearing, perhaps an air bearing system, might find application in transmission systems. We are, after all, talking 21 years in the future.

Along with other than shaft-driven heavy lift helicopters, of course, we would also benefit from the development of a precision hover system. And, if you want good fidelity of a flight control system, development of an all electrical flight control system is in the cards. I have shown here (Figure 12) possible flight control systems for a VHLH, a mechanical system of course is used in most present-day helicopters. The electrical system, with a mechanical backup, is a potential applicant, and the all-electrical system with proper redundancy and proper proven reliability is also a candidate. The all-electrical system goes to save a few tons in the control system weight in a VHLH and it ought to be developed for this time period, perhaps even prior to this time period. The electrical system has other benefits, as well. Mechanical system is limited as to exactly how mixing is put in or taken out or how slop is kept out of the system. Every bolt, every joint, has a certain amount of slop. Every flexure - every component in the aircraft that moves or flexes-gives you a certain amount of hysteresis. In an electrical control system a great deal of this might be taken out, and hover precision, reduced weight, and the same or improved overall reliability also result. I say electrical system, but you notice we still have the pilot in there and he's still handling mechanical controls. At the other end, in this case, we still have hydraulics in there and the muscle is still provided by a hydromechanical system. It is the system between that would be different in an electrical control system. To me, flight controls changes are one important means of keeping the weight empty fraction low on the VHLH of 1990. As with any variety of VHLH, whether shaft-driven or tip-driven, the cargo handling system needs to have a great deal of attention paid to it, and the aircraft technology, in general, does need to have attention paid to it, again to keep weight fractions down and to keep the life and reliability of the systems up. That is my story on shaft-driven VHLH technology in 1990. (Figure 13.)

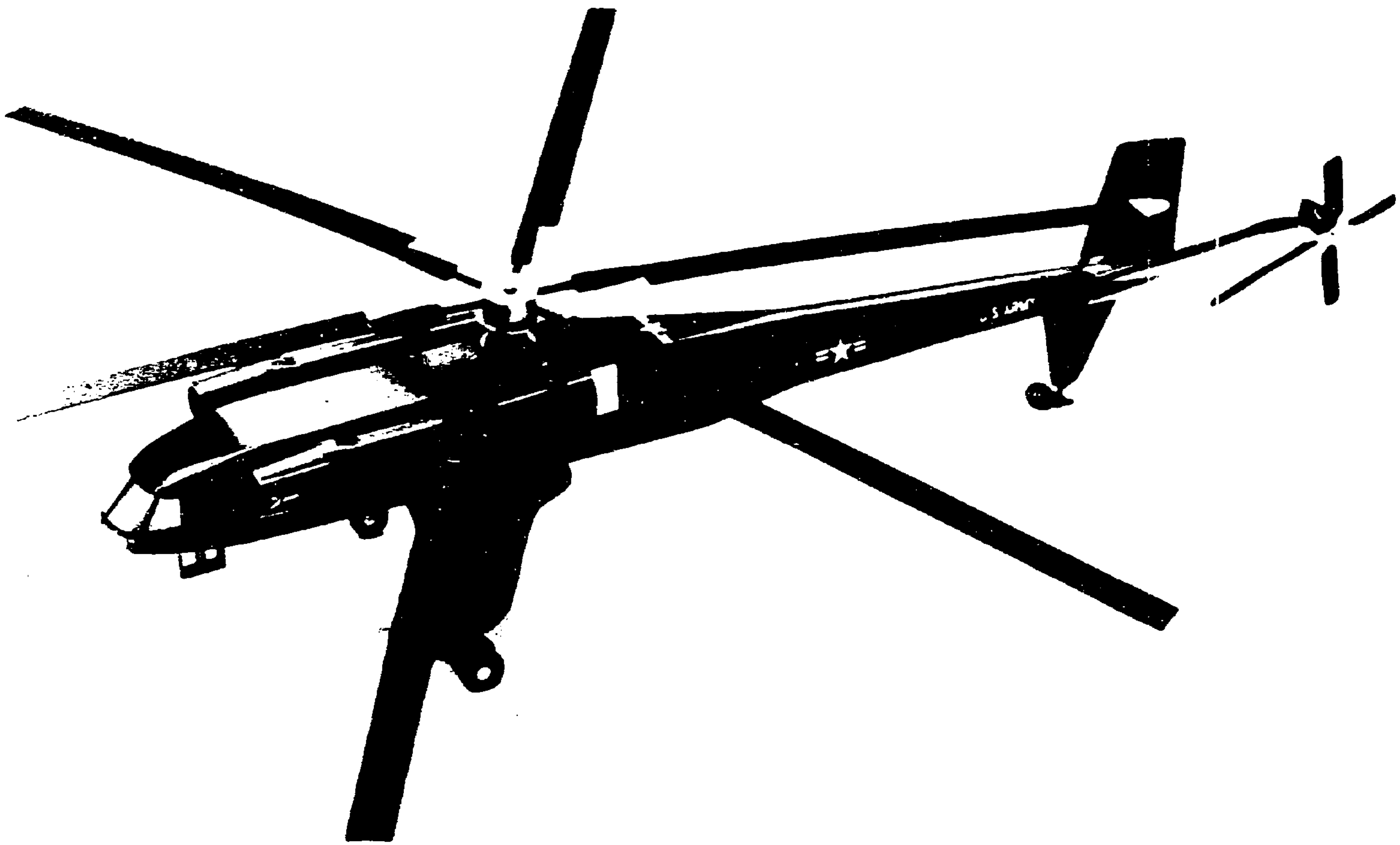


FIGURE 1

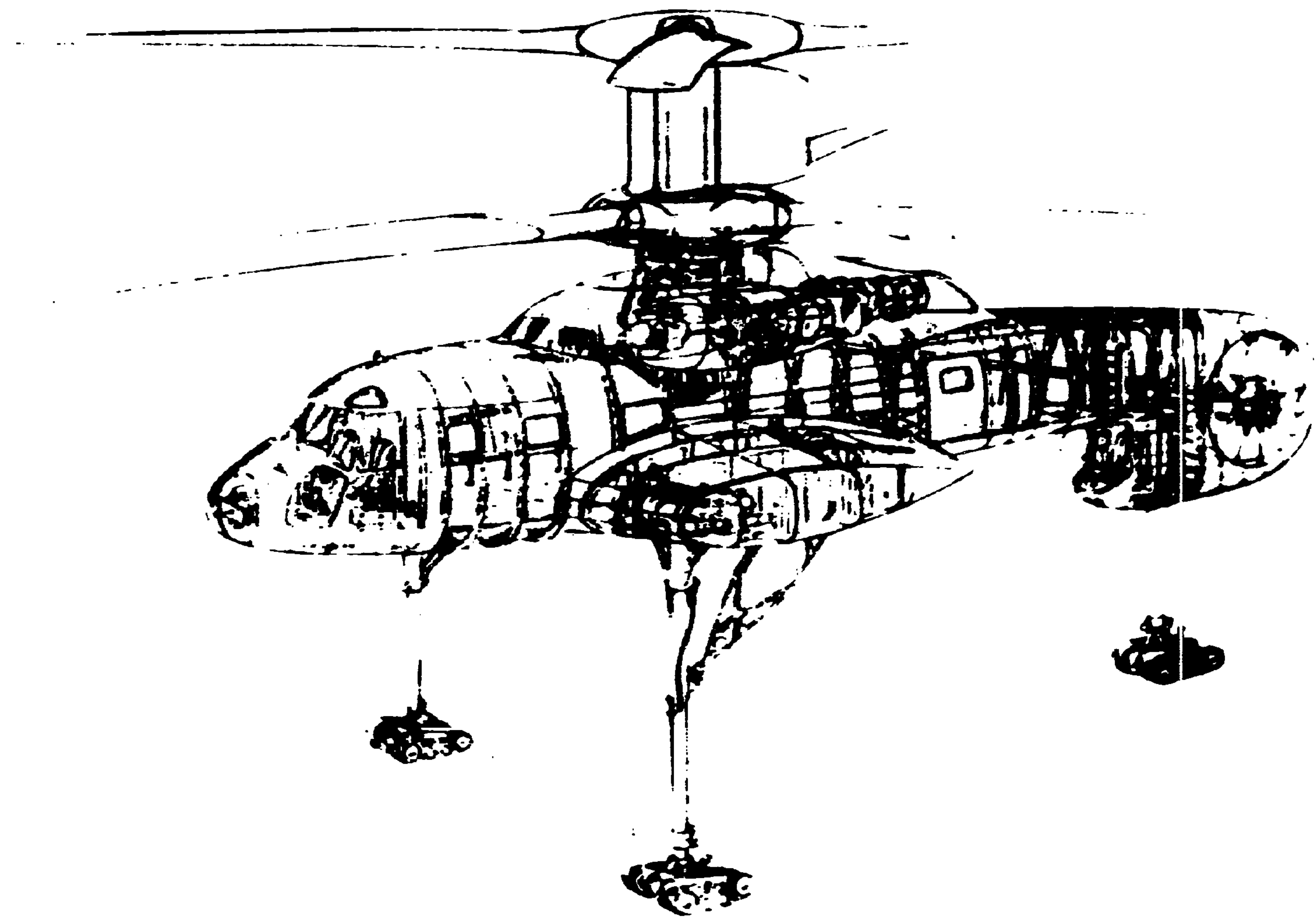


FIGURE 2

2OO-14

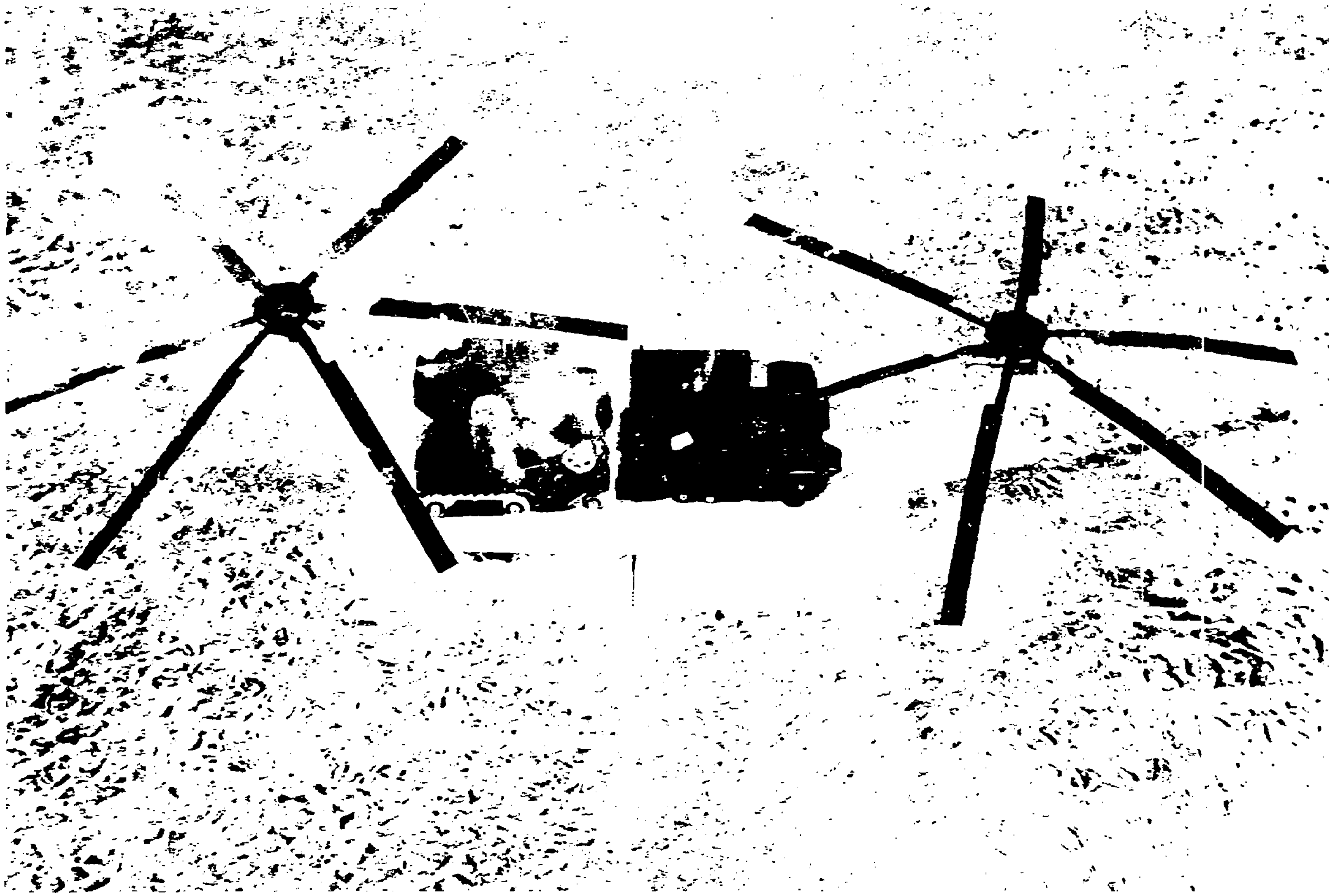


FIGURE 3

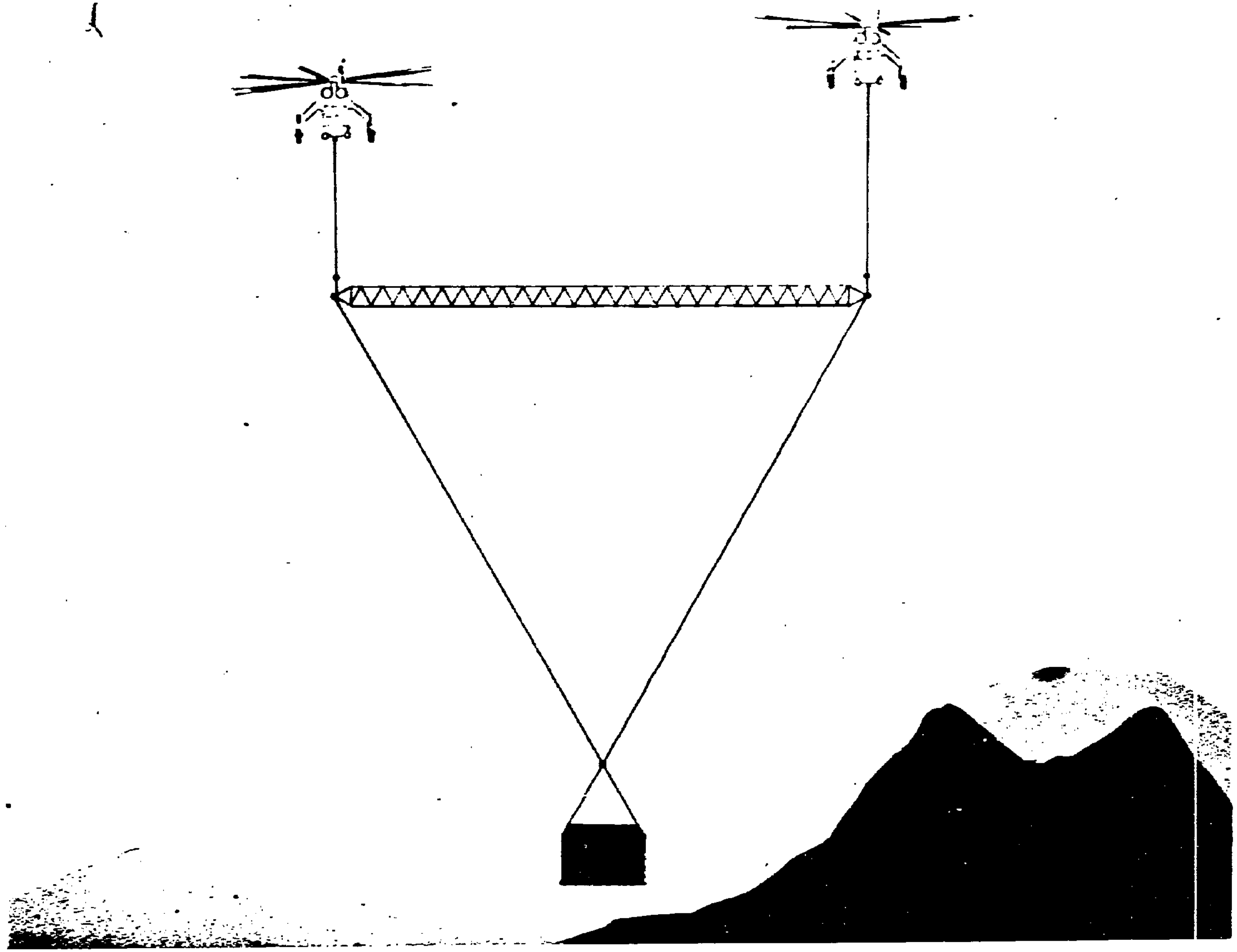


FIGURE 4

IMPACT OF TECHNOLOGY ON PAYLOAD FRACTION

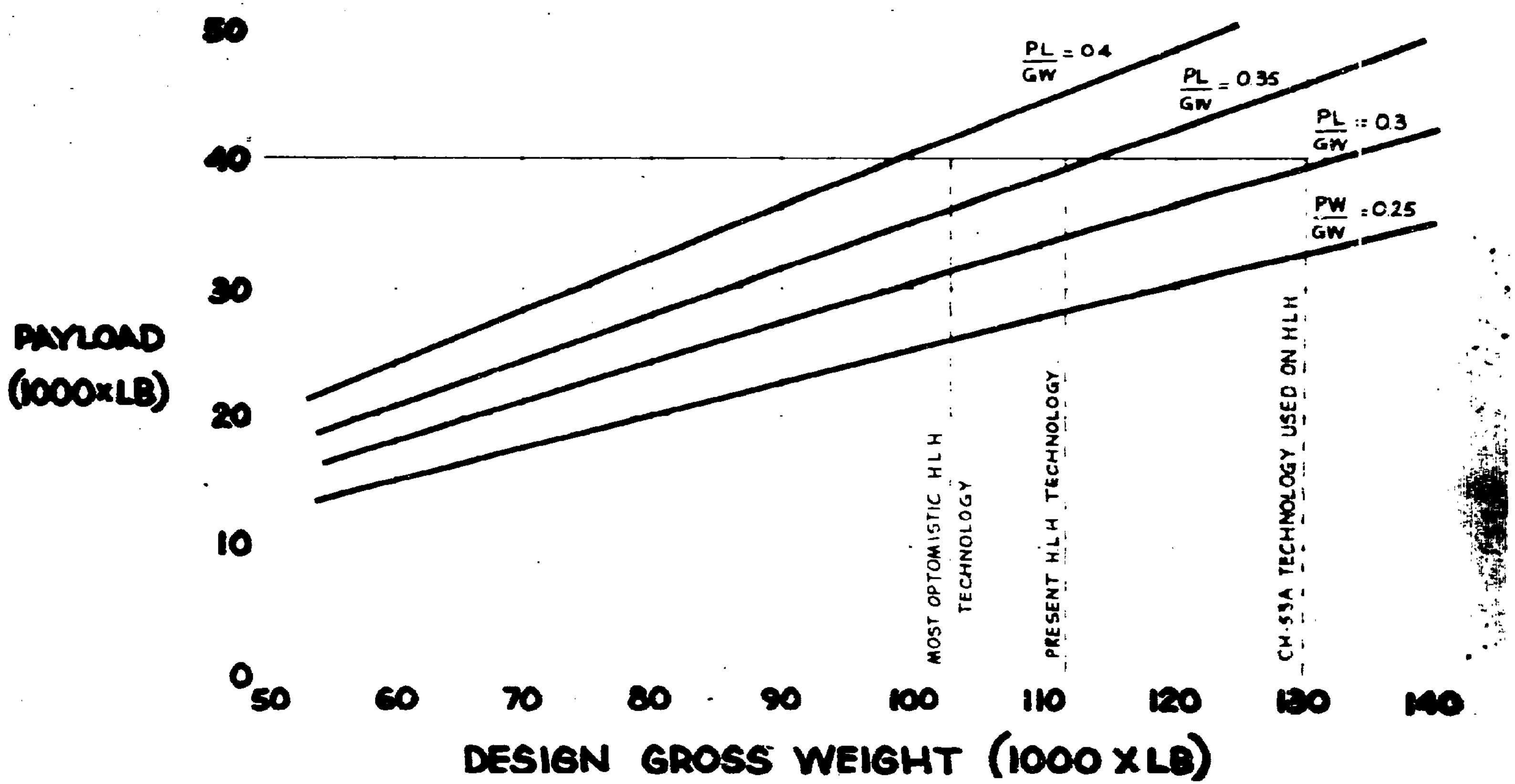


FIGURE 5

HLH
ROTOR BLADE MATERIALS
27 TON PAYLOAD, 69 TON GROSS WEIGHT
4 BLADES, 140 FOOT DIAMETER

BLADES	WEIGHT LBS.	F_C • LBS.	LIFT • LBS.	CONING ANGLE DESIGN G.W.	CONING ANGLE ALT. G.W.
AL. ALLOY	3,400	420,000	138,000	4.2°	5.2°
TITANIUM	2,300	276,000	138,000	6.7°	8.4°
COMPOSITE	1,375	180,000	138,000	11.0°	14.1°

FIGURE 6

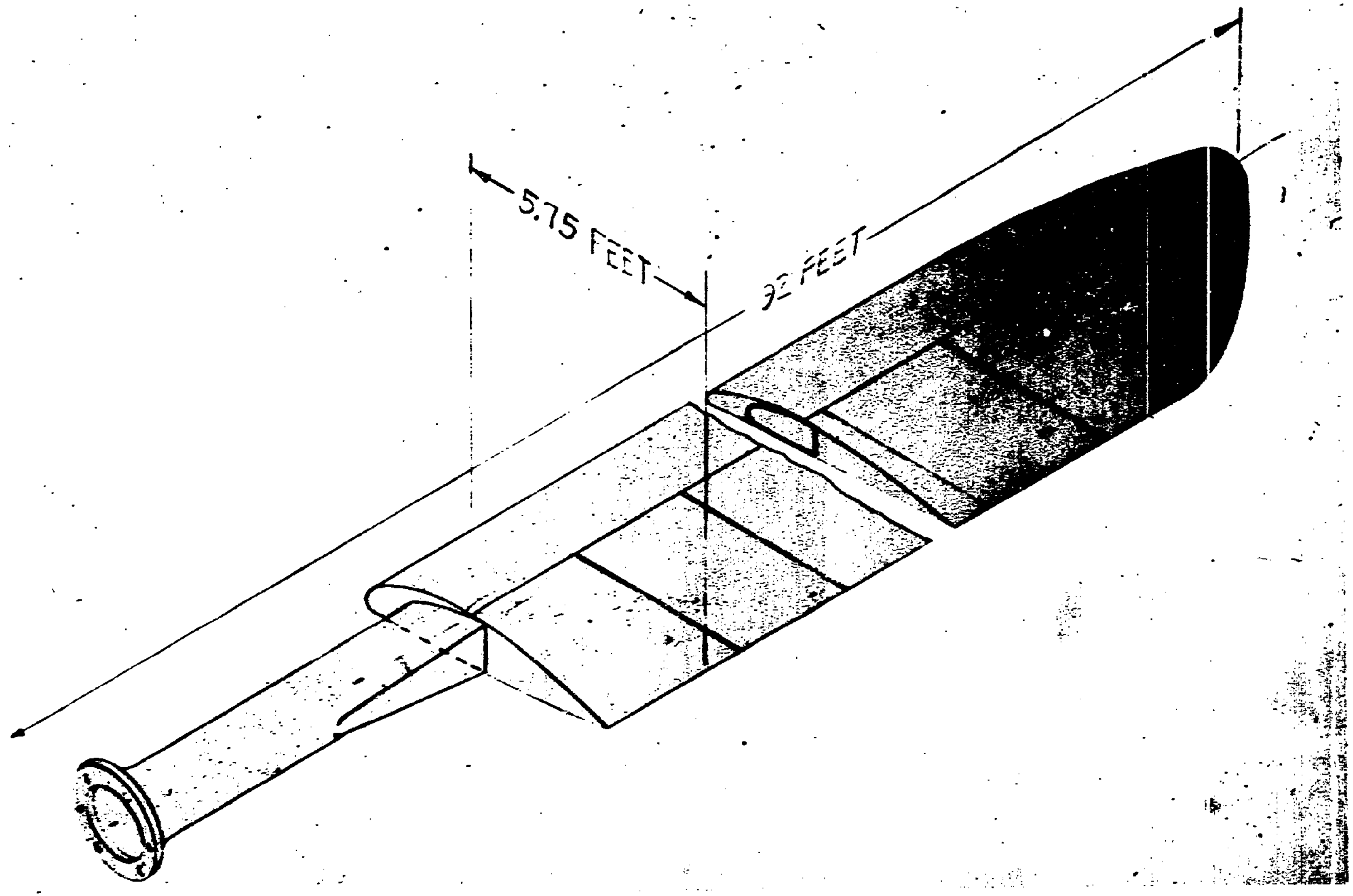


FIGURE 7

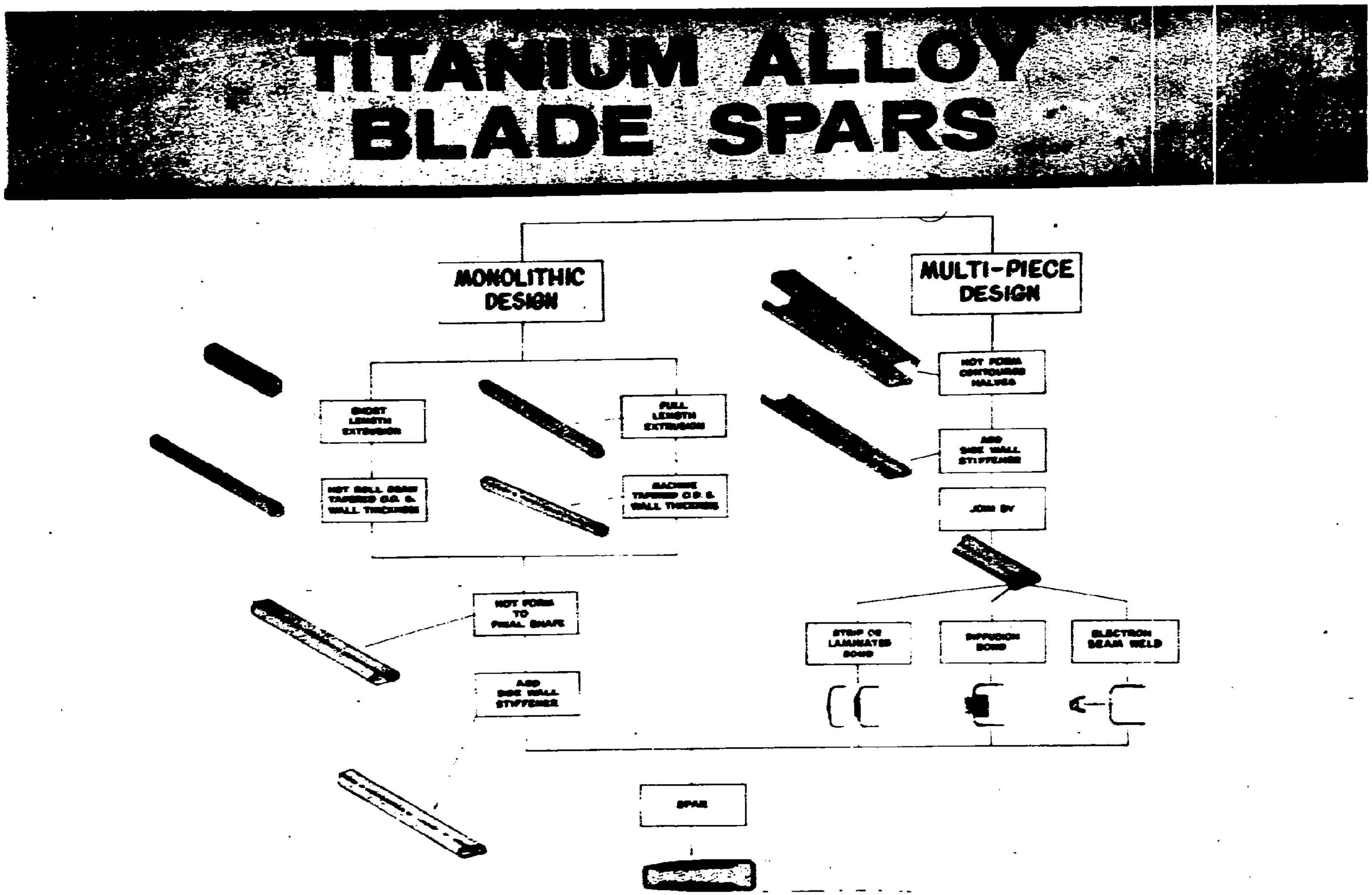

TITANIUM ALLOY
BLADE SPARS
MONOLITHIC DESIGN
MULTI-PIECE DESIGN

FIGURE 8

ROTOR HEAD ASSEMBLY

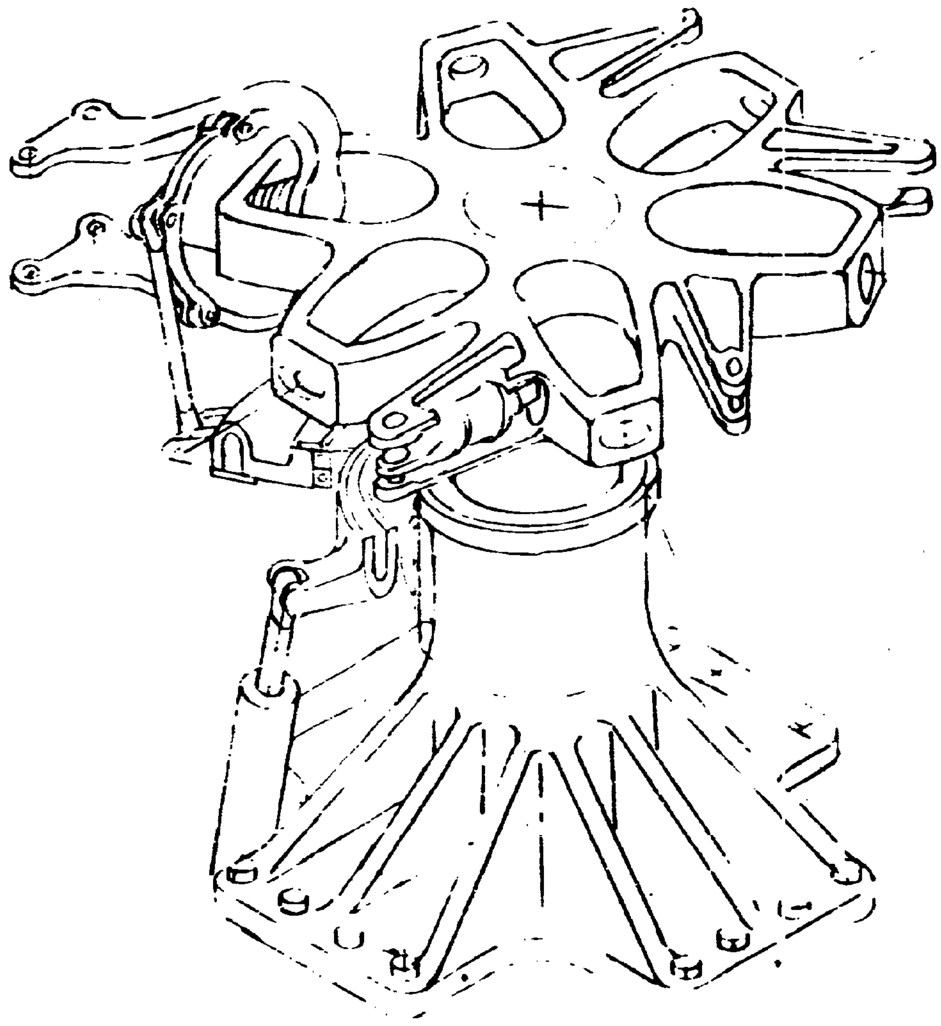


FIGURE 9

SPHERICAL ELASTOMERIC BEARING TESTS-1963

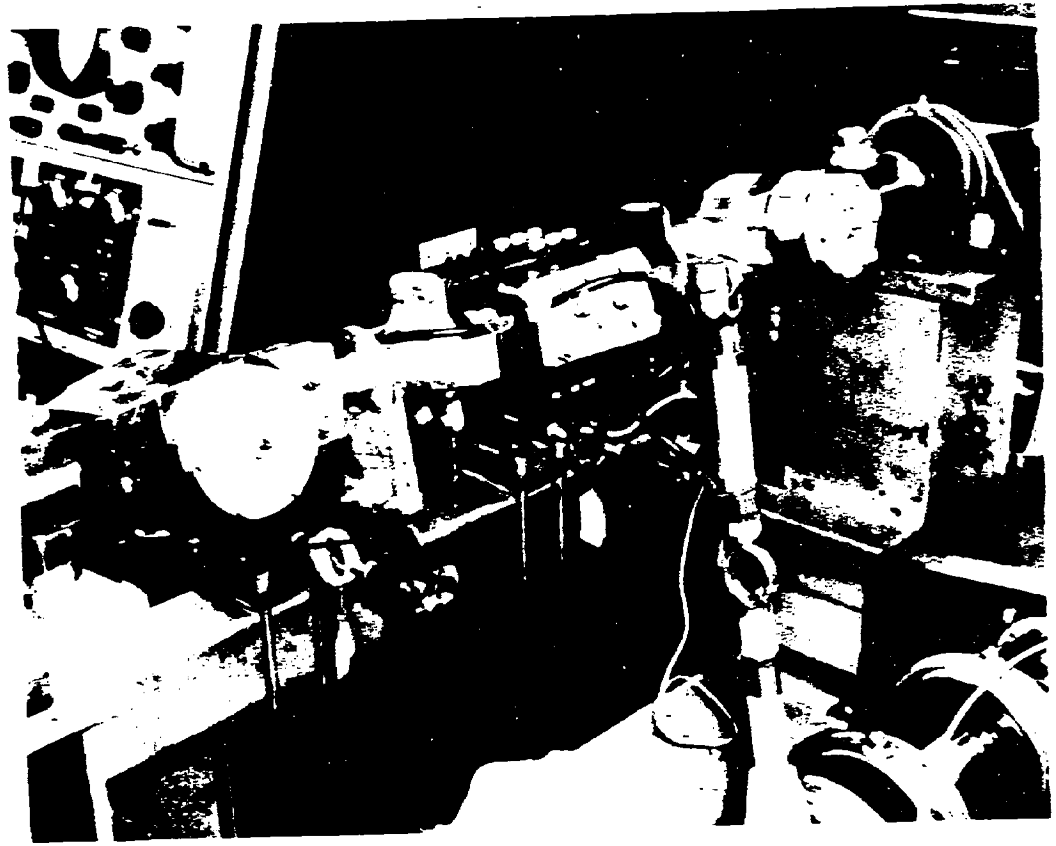
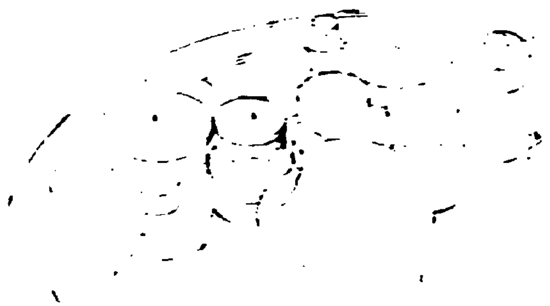
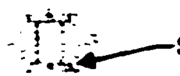


FIGURE 10

HLH GEARING FEASIBILITY STUDY



ROLLER GEAR DRIVE



ADVANTAGES:

- LARGE SPEED REDUCTION IN MINIMUM HEIGHT (REDUCTIONS STACKED RADIALLY INSTEAD OF VERTICALLY.)
- BEARINGS REQUIRED ONLY IN FINAL PLANET PINION ROW.
- LOW GEARBOX WEIGHT IF LARGE REDUCTIONS CAN BE UTILIZED.

DISADVANTAGES:

- LARGEST UNIT FABRICATED *TO DATE IS A 1,000 HP UNIT.
- LOW REDUCTION RATIO UNIT - NO WEIGHT ADVANTAGE OVER CONVENTIONAL PLANETARY
- LARGE REDUCTION RATIO UNIT REQUIRES RING GEARS LARGER THAN CAN BE CURRENTLY MANUFACTURED.

* CURRENTLY UNDERGOING PRELIM. DEV. AT TRW. INC.

FIGURE 11

CANDIDATE CONTROL SYSTEMS

I. MECHANICAL

SAS

PILOT ⊕ MIXER SERVO ROTOR

II. ELECTRICAL WITH MECHANICAL BACKUP (HYBRID)

SAS

MIXER

PILOT SERVO ROTOR

SERVO MIXER

III. ELECTRICAL

SAS

PILOT MIXER SERVO ROTOR

MECH. HYD. ELECT.

FIGURE 12

FIGURE 13

APPENDIX J
TECHNOLOGY BACKGROUND, FANJET DRIVE HLH

Presentation by Mr. K. Amer - Hughes Tool Company

The tip drive side of the heavy lift picture is going to be presented in two parts. The technology side (Figure 1) of the story will first be presented and then the results of a cost effectiveness study will be shown. Our system is called "Fanjet Drive" which is a somewhat new name, and it is basically related to the advancement from the earlier hot cycle concept to the bypass engine concept, and the bypass engine is sometimes known as the fanjet engine.

Figure 2 is a 3-view drawing of one possible fanjet-drive or warm-cycle-type of very heavy lift helicopter. Some of the key features are:

1. The elimination of the main transmission

2. A considerably shorter fuselage as compared with the single-rotor shaft-drive configuration, which would require a fuselage to go beyond the diameter of the rotor and

3. A very small tail rotor or fan for maneuver control as opposed to the rather large tail rotor that would be required in a single-rotor shaft-drive configuration.

These savings lead to weight savings and this, of course, is the fundamental reason for being interested in the tip-drive configuration. The intent is that sufficient weight savings can be achieved so that the payload can be carried in a considerably smaller vehicle. The next few figures give some idea of what heavy lift helicopters or very heavy lift helicopters would look like as a function of payload. It would be well to interject here that we are using about 1975 technology. It is certainly felt that 1975 is not too near to talk in terms of a 30-ton capability, and for a 50-ton payload, it is felt 1980 is not too early. In looking at shaft-drive technology, larger gross weights and larger rotor diameters will be seen than those Lou Knapp of Sikorsky was talking about (but of course, he was talking 1990 technology). Also, in all honesty, we do not know as much about shaft drive as Sikorsky or Vertol, or conversely, they do not know as much about tip drive as we do, so invariably there is probably a degree of pessimism when looking at the competitive side.

Figure 3 shows a ball park estimate of gross weights versus payload requirement. It is recognized that, in addition to payload, there are questions of mission radius and special equipment that may or may not have to be carried that also influence the gross weight. Just to get a ball park idea, you can see that in going from maybe 20 tons to 50 tons for shaft drive (again 1975 technology), the weight would go from a little over 100,000 pounds up to about 300,000 pounds. With jet drive you could anticipate savings in transmission, fuselage, tail rotor, and so forth, and the weight would get to about 200,000 pounds in the 50-55 ton class. It is worth noting that in any payload requirement, whether it is 30 or 40 or 50 tons, there is a significant difference between shaft drive and jet drive in the gross weight required to fly a mission, or conversely, if gross weight is considered to be a level of technology which is limiting at any particular time frame for any given level of technology, you can get about 10 more tons of payload in the jet drive. This anticipated superiority of the tip drive is the basic reason for interest in it and for our study of it.

A similar investigation looking at rotor diameter is shown in this next slide (Figure 4). This chart was based on the previous one and is based on a disc loading of 10 lb/sq. ft. I would like to make a short comment on disc loading relative to the square/cube law that we have been hearing about. One of the means of delaying the penalty of the square/cube law in an airplane would be to go up in wing loading. The equivalent step in a helicopter would be to go up in disc loading. We are now reaching the point in helicopter development where we are just about 10 lbs/sq. ft., and now we have to worry about downwash velocity. Well, rightly or wrongly, I have used 10 lb/sq. ft. as the disc loading limit in this study. Using that as a limit, the shaft drive goes from about 120 feet diameter in the 20-ton category to about 200 ft. diameter for roughly a 50-ton payload and again, at any given payload requirement, the jet drive can do the mission for a significantly smaller diameter. Alternatively, if rotor diameter is considered to be a measure of technology, and of the limit in state of the art, you can get about 10 more tons in the jet drive than in the shaft drive.

Figure 5 shows what is called rotor torque ratio. This is the variation of rotor shaft torque using a 10-ton payload machine as a reference of one. The 10-ton payload is about the present state of the art in helicopters. This chart shows a factor of roughly 16 in shaft torque to get to a 50-ton payload. As I recall, Lou Knapp of Sikorsky was using a number like 12 as a ratio, so we are not too far apart. But, this does give you an idea of the magnitude of the problem in trying to do the 55-ton payload job in the shaft-drive configuration. Again, this is another one of the reasons why we are so interested in tip-drive.

Another item of interest is to plot rotor diameter against first flight date (Figure 6). All of the black circles are shaft-drive helicopters. The largest rotor diameter of a shaft-drive American helicopter was the YH-16, which was a little over 80 feet in diameter. The Mil-10 was about 100 feet in diameter and the Mil-12, we understand, uses the same rotor. So, all these black circles represent shaft drive.

The XH-17 was, of course, a tip-driven helicopter which flew back in the early 1950's. Its rotor was 130 feet in diameter. As can be seen, it was way ahead of any shaft-drive helicopter in its time, and to this day, as far as we know, it still represents the largest diameter rotor that was ever flown either here or in Russia. This indicates that there appears to be an advantage in the tip-drive and any level of technology as represented by rotor diameter appears to be easier to get in a jet-driven rotor than with a shaft-driven rotor. So the previous charts show that to do the mission with a jet-driven rotor requires a smaller diameter rotor, but even at equal diameter rotors, it appears to be easier to get flight hardware with tip-drive.

Next is a photograph of the XH-17 (Figure 7). Again, it shows the small length fuselage and a very small maneuvering tail rotor. Keep in mind this helicopter flew 20 years ago.

Figure 8 is the XV-9A which was a hot cycle research helicopter. In this propulsion system the hot gases from the jet engines were ducted up to the rotor head, out the blades and exhausted through nozzles at the blade tips. The big dome on the rotor head is instrumentation.

The next chart (Figure 9) is from preliminary studies of a fan-jet-drive helicopter in the 30-ton payload class. The two main points that we want to show with this figure are that using a 4000 ft. 95°F hover criterion, we estimate about 127,000 pounds gross weight to achieve 30 tons of payload and, as we change the hover criterion, the same helicopter at sea level and 95°F could go up another 20,000 pounds in gross weight and in payload. In other words, you can carry another 10 tons at sea level 95°F. The 4000 ft. 95°F condition has a much higher percentage probability. Therefore, we could carry the additional 10 tons of payload for perhaps half or 60 percent of the time and only under relatively less likely occurrences would you be restricted to the 30 tons of payload. Under sea level standard conditions, another 7 tons or so can be carried.

A point worth noting is that there is some discussion about flat rating or derating requirements. In the tip-drive configuration, there is no need for derating. In other words, a jet engine puts out a certain amount of power under a hot day condition at full throttle or at limit turbine temperatures. Under less stringent ambient conditions at the same engine turbine temperature, the engine exit gases are more dense and, therefore, there is a higher tip thrust. There is no structural limit to pulling the full power of the jet engine under less stringent ambient conditions so there is no question of derating. The only requirements are that the design load factor be sufficient at the basic design point to permit the higher gross weight at sea level and that adequate strength be provided in local areas such as cargo handling equipment. Alternatively, you might consider for example, a 30-ton capacity winch, while the 40-ton or 47-ton payload capability would be satisfied with just a fixed sling. It might be mentioned that, for example, on the Mil 10 there is no hoist. All cargo on the Mil 10 is carried by a simple cable. So these are possible areas of flexibility in operation in which a vehicle could carry from 30 to 47 tons, depending upon the required hover performance.

Figure 10 summarizes the advances that have taken place over the past 15 to 20 years in the area of tip propulsion. In the first column we have the different configurations starting many years ago with the ramjet and pulsejet, advancing to the tip-burning pressure jet such as on the XH-17, the cold-cycle pressure jet which is used on the French Djinn helicopter, the Hot Cycle XV-9A Army/Hughes Program, and the lastest work in fanjet hot cycles using bypass engines. Notice how the sfc which in the ramjet/pulsejet days was 5 to 10 has improved to a value of about 1 on the XV-9A, about .85 using currently-available engines and predicted improvements of .78 using advanced bypass engines currently under development. It might be pointed out that these are sfc's in pounds per rotor horsepower-hour, so these numbers can not be compared directly to shaft engine sfc's.

Other points of interest in this figure are that the XV-9A used gases of about 1200°F so that we have experience in the XV-9A designing ducting seals and so forth to accommodate 1200°F gases. The new studies are based on bypass engines which, of course, mix cold air with the core's hot exhaust and we are now dealing with roughly 800°F gases which clearly is a much simpler design problem than what has already been solved on the XV-9A. It is also worth noting this last column on jet velocities. The tip-burning pressure jet for example is a very noisy system. Those of you who are familiar with the Rotodyne program in England remember that the Rotodyne was quite a successful program that was killed by the noise problem. But notice that the reason for the noise was the jet velocity of about 3000 ft/sec. The hot cycle on the XV-9A was just a little over 2000 ft/sec. The bypass engine jet velocities are in the neighborhood of 1900 ft/sec. The tip jet noise varies roughly as the eighth power of jet velocity. The difference, therefore, between the early tip burners and the modern bypass cycle is just like the difference between an afterburning and non-afterburning jet airplane. These major improvements in noise have reached a point where we do not anticipate that jet velocities will be a source of significant noise problem.

Next is a chart prepared to compare sfc for the fanjet drive using the warm cycle with the shaft drive (Figure 11). The comparison is made at both 100 percent of military rotor power and at 50 percent of military rotor power. Most of the operation would be at roughly 50 percent of power. This is because cruise and the portion of hover when you are not carrying payload would be at 50 percent power, especially if you design for a hot day altitude capability and you are doing most of your flying at more temperate temperatures.

Sfc based on gas horsepower is used in order to have a common denominator. Sfc based on gas horsepower is fairly close for the two configurations, with the slight advantage in the fanjet drive because of the fact that the fan is supercharging the core engine. The same core engines are assumed in both cases. The engine horsepower sfc are put in for the shaft drive. Then shown are the ratios of main rotor horsepower divided by gas horsepower. A value of .424 is the number for the fanjet drive, while .705 is the number for the shaft drive. Now, sometimes people mistakenly compare this ratio for the tip drive with a value of 1.0 for the shaft drive. Of course, that is not the case. There are losses in the shaft-drive configuration when you start with gas horsepower due to power turbine efficiency, leaving loss from the exhaust of the engine, tail rotor, and transmission. Applying the ratio of main rotor horsepower to gas horsepower for the jet drive to the basic gas horsepower sfc you get an sfc based on main rotor horsepower which in this particular case is .821. It is a little higher than the previous slide because the previous one did not include the yaw control and accessories which are now included in both of these numbers. Notice that the number for the shaft drive rotor sfc is a little higher than 0.5 even though it started off with a basic engine sfc of 0.43. The final line is the ratio of sfc's between the tip drive and the shaft dirve, which is 1.62. So the jet drive has a 62 percent higher sfc at 100 percent power. Now, when we go to 50 percent power, which is the power level that really counts because that is where you do most of your operation, the comparisons are basically similar except now the ratio of sfc is 1.48. Now, it may be asked how do we go from 1.62 sfc ratio down to 1.48? The answer is basically shown on this line where the ratio of main rotor power to gas horsepower for the jet drive improved from .424 at 100 percent power to .483 at 50 percent power. This improvement is basically associated with the tip propulsion efficiency, where at lower powers we have a lower tip-jet velocity and, therefore, we have a better match between tip-jet velocity and rotor tip speed. So we have typically about a 1.5 penalty in sfc, compared to the shaft-drive configuration.

Now, what does this sfc ratio mean in terms of mission fuel requirements? This bar chart (Figure 12) answers this question. (Depending upon which level of the state of the art is used, and depending on which company does it, you get somewhat different numbers but the basic concept is this.) A mission with a specified payload is assumed. Here is a shaft helicopter and here is a jet helicopter, both with equal payload requirements. Now, what I am showing here is essentially equal fuel quantities. You will immediately say, how can that be since our sfc's are 50 percent higher? The answer is basically here in the airframe weight. The elimination of the transmission, the shorter fuselage, the elimination of the tail rotor, and the fact that the fuselage is not subjected to tail rotor loads -- all these permit significant savings in airframe weight. The net result is that the gross weight of the jet helicopter is significantly less than the gross weight of the shaft helicopter. The fuel quantity required for a mission is based on power required times the sfc. Now, with the lower gross weight of the jet drive you have lower power required times the higher sfc. So the fuels for the mission are very similar for the jet drive and the shaft drive. If you pin me down as to whether the fuel quantities are exactly equal or whether our fuel is 10 percent higher than the shaft-drive helicopter fuel -- I couldn't answer that question. The main point is that the difference in fuel is not 50 percent just because the sfc differences are 50 percent. The fuel difference is more like 10 percent or maybe even equal, depending upon the comparative airframe weights.

Now, how do we know that the sfc of our system is only about one and a half times the shaft-drive sfc? Well, this is our background. In order to establish sfc, we conducted many propulsion analyses (Figure 13). The key parameters are as follows: we start with engine exhaust temperature and pressure; that is, the temperature and pressure of the gases coming out of the tail of the engine. We want to establish the blade pressure change from root to tip and that means that you have to look at the duct friction and we have to look at the effect of centrifugal force acting on the gas in the blade. Finally, a reasonable nozzle velocity coefficient must be established. This is a measure of the efficiency of converting gas pressure ahead of the nozzle into jet velocity and, of course, the efficiency determines how much of a momentum change, and therefore, how much of a tip force you get. Well, in order to establish these parameters a tether test was run on our XV-9A helicopter.

The XV-9A not only gave us an opportunity to demonstrate the feasibility of the hot cycle system it also gave us an opportunity to measure some fundamental engineering numbers. We ran a tether test (Figure 14) where we restrained the three blade tips with load cells. We also ran pressure taps into the blade ducting. We ran the engines at various power levels up to full power, measured the tip thrust, and measured the pressure change along the blades.

Now, this (Figure 15) is an example of how we worked up the data. Here is a plot of the nondimensionalized local duct pressure against blade station and these symbols represent experimental data points. Here are three theoretical curves with friction coefficients of .004, .003, and .002. As you can see, the experimental data pretty well matches the .003 skin friction coefficient. On this basis, we have established what we feel is a valid friction factor of .003. It is anticipated that in future larger machines with higher Reynolds numbers, this number should come down some. However, thus far we are not taking advantage of that. A value of .003 was used in all of our propulsion analyses. Once you know the drag coefficient, you can predict the drag for any combination of blade chord, ambient conditions, density, and so forth. So, similarly, knowing the appropriate duct friction coefficient we can now predict the pressure loss in any size duct, any length duct, with any gas velocity.

Figure 16 is the data that was plotted up to establish the nozzle velocity coefficient. The vertical parameter is "F", the tip force measured during the tether test, divided by some gas parameters. The horizontal parameter is the total pressure just upstream of the nozzle ratioed to ambient pressure. The data points represent experimental points measured during the tether tests. Again, a comparison was made with theory. Shown is the 1.00 line, the 0.95 line and the 0.90 line. The dashed line, which is the theory for a value of velocity coefficient of 0.94, pretty well matches the experimental data. By this means it was established that on the XV-9A, we had a nozzle velocity coefficient of 0.94. Now, as frequently happens on limited budget research programs, some compromise was made in the design of the nozzle. We feel that a realistic objective is 0.96 for a nozzle velocity coefficient, and that is the number that was used in our propulsion analyses for the HLH configuration.

Before concluding the propulsion part, I think a word about engine availability is important. In Figure 17 we compare fanjet-drive engines and shaft-drive engines in two categories: available today and estimated availability by the mid 1970's. Now with three engine companies' representatives in the audience, I am a little nervous at the moment, but this is my interpretation of what we have been told by various engine company representatives. With regard to engines available today, the Allison TF-41 is in production and is scheduled to be used on the LTV/Air Force A7D and Navy A7E. So, it is an engine that is in production today and makes quite a good powerplant for a warm cycle heavy lift helicopter with no modification. This means: (1) the engines are available, (2) there is no R&D cost for the Army, and (3) it means the Army gets an engine that is completely debugged by the time the engine is required. The Army does not have to go through the pains of the debugging program, setting up maintenance procedures, etc. That is all done. Conversely, for the shaft-driven helicopter configuration, there are no large free-turbine turbo-shaft engines in production today.

Now, looking into the future to see what will be available by the mid-1970's. There is the Pratt & Whitney JTF10A, an engine currently being developed for the F111D. The F111D is, of course, an Air Force airplane. In addition, there are two engines, the Pratt & Whitney JTF22 and the General Electric GE1/10F10 that are in competition for the F14B Navy airplane and the F-15 Air Force airplane. Sometime before the end of the year it is presently planned that one of these will be selected and carried on into development. So here essentially are three engines that are being developed with Air Force and Navy money, and again, they would be available for Army use by the mid-1970's with essentially no expenditure of Army R&D funding.

In the shaft-drive engine class, this first engine is a conversion of the J52 and this second engine is a conversion of the GE1. Based on what the engine companies are saying today, these engines can be converted to turboshaft engines by the addition of a power turbine. There is no question that a turboshaft conversion of these engines is feasible. The only question is whether the $30 or $40 million is available to do the job.

One point is worth noting. There is a fundamental reason why the fanjet drive has four engines to choose from -- one available and three in development, while the shaft drive has none available and would require a specific Army development program. The answer is basically that there are requirements in the fixed-wing airplane industry for fanjet engines, and by a fortunate coincidence the optimum bypass ratio for fighter airplanes is about 0.8 and the optimum bypass ratio for a warm cycle is about 0.8 and so with no conversions we have available four engines -- four bypass engines that are essentially ideal for the warm cycle. Conversely, there are no fixed-wing airplanes using large free-turbine turboshaft engines, and therefore, a specific Army-funded development program would be required.

Now, moving on into the structures area, Figure 18 is a cross-section of the blade on the XV-9A. Two ducts carry the 1200°F gases. There are two spars, with an air space that acts as insulation to prevent the hot temperatures from reaching the spars. The spars were of steel and, as shown in Figure 19, the temperatures were low enough that there is no penalty in material properties. In addition, some cooling air was allowed to be pumped out in these two spaces by centrifugal force. An aft segment completes the airfoil section.

This chart (Figure 19) compares measured temperatures versus predicted temperatures in the XV-9A design. In each case the predicted temperature is underlined and the adjacent number which is not underlined is the measured temperature number obtained during flight tests. During flight test temperatures of the duct skin, the outer skin, the spar cooling air were measured. The predicted temperatures are based on three-dimensional analysis of chordwise, vertical and spanwise heat flow. This figure shows good agreement between predicted and measured temperatures, which give us confidence in being able to predict temperature distribution on any future design, and thus be able to select the proper materials.

Another area is one of duct seals (Figure 20). There were 4 primary seal areas on the XV-9A. The rotating seal was on an annular duct so it had an outer and inner rotating seal. Then there was what is called the articulate duct at the blade root which again had an inboard and outboard seal which permitted the blade articulation. All four of these seals were designed for 1200°F. Today, if these seals were designed for 800°F, it would be a lot simpler job. There was no particular trouble in any of these seal designs.

Now, Figure 21 is a picture of the outer rotating seal. Shown is the rotor shaft, and part of the seal that rotates. It is flame-plated tungsten carbide and a series of segmented carbon graphite segments contacted this surface to permit the rotation. The concept was actually not original with us. This was a seal concept that has been used in the engine industry. Our contribution was that this seal was (and believe it still is) the largest rotating seal of its type. There were no particular problems with this seal.

Now, this next figure (Figure 22) summarizes what was learned on the XV-9A program. At the end of the program a pressure test on our duct and seal system was run all the way from the engine tailpipe to the tip nozzle, and there was negligible gas leakage. This was after about 160 hours of total operation which included about

35 hours of flight operation. The leakage was about 2/10 of one percent. There are no seal problems and no leakage problems. Coming down the list, we had 160 hours of satisfactory operation -- the predicted hover performance was verified. Although it was a limited budget program with the resulting penalties in weight, a useful load over weight capability of about 1 was achieved, in combination with a hover ceiling of 6000 feet 95°F, if we had been given production T64 engines. The actual engines were prototype "Y" engines but if we had production engines and with minor modifications, a hover ceiling of 6000 feet 95°F could have been achieved. There were no temperature problems, good blade dynamic characteristics, the noise level was acceptable and the maintenance level was quite acceptable. So this program does give us confidence in our understanding of the concept, and as was mentioned before, it did give some hard engineering numbers that are needed for future performance prediction.

Now, as a final item, in talking about weight savings we are sometimes questioned on whether we have a good basis for predicting weight. It would be best to say that in predicting the weight of a large tip-drive VHLH we are using engineering judgment just as the shaft-drive proponents are.

TECHNOLOGY BACKGROUND

Fanjet Drive HLH

FIGURE 1

Revised 8 May 1969

HLH 55 TON PAYLOAD DESIGN CANDIDATE

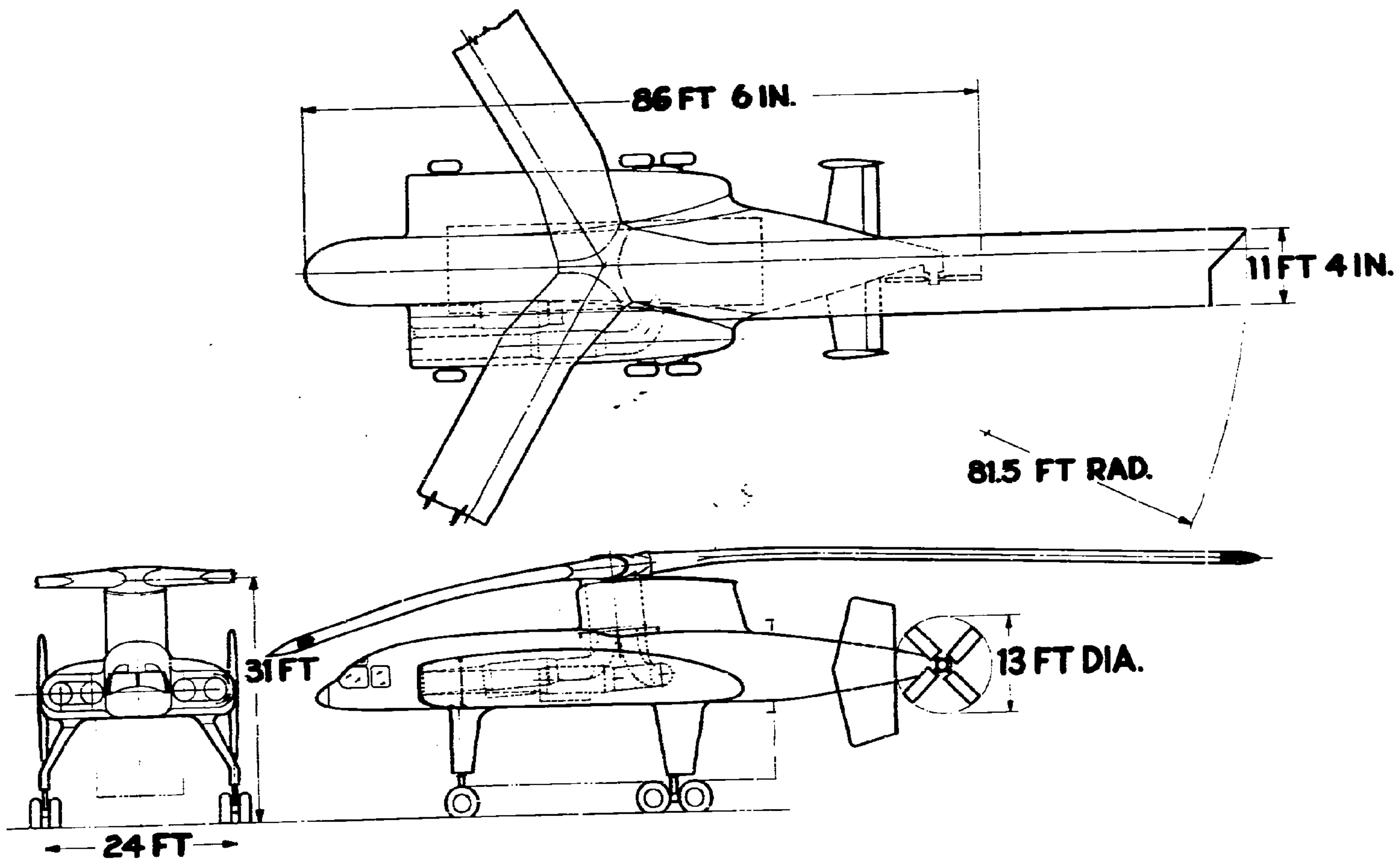


FIGURE 2

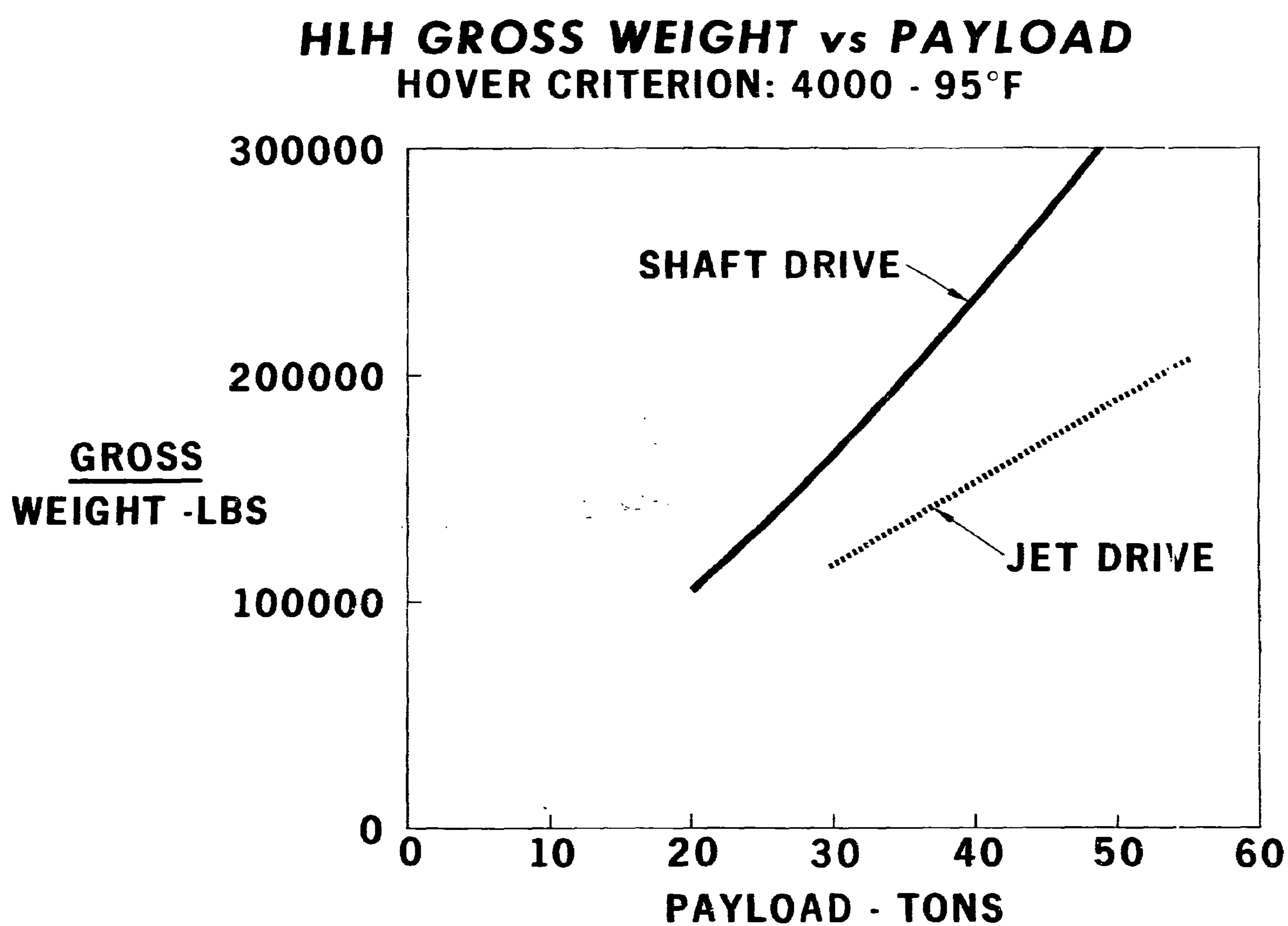

HLH GROSS WEIGHT vs PAYLOAD
HOVER CRITERION: 4000 - 95°F
GROSS WEIGHT -LBS
300000
200000
100000
0
SHAFT DRIVE
JET DRIVE
0
10
20
30
40
50
60
PAYLOAD - TONS

FIGURE 3

HLH ROTOR DIAMETER vs PAYLOAD
HOVER CRITERION : 4000 - 95° F

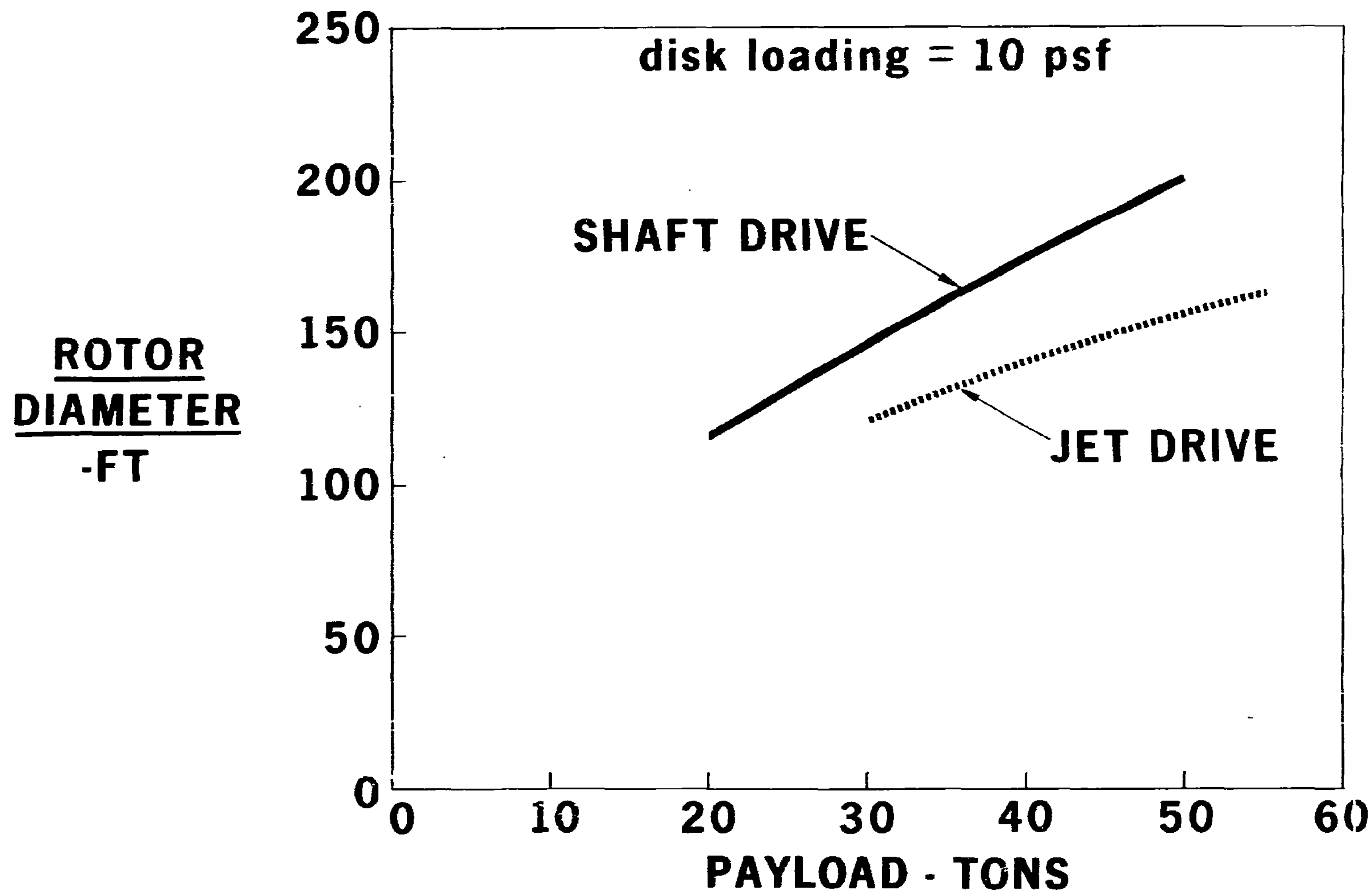


FIGURE 4

ROTOR TORQUE RATIO vs PAYLOAD
SHAFT DRIVE HLH

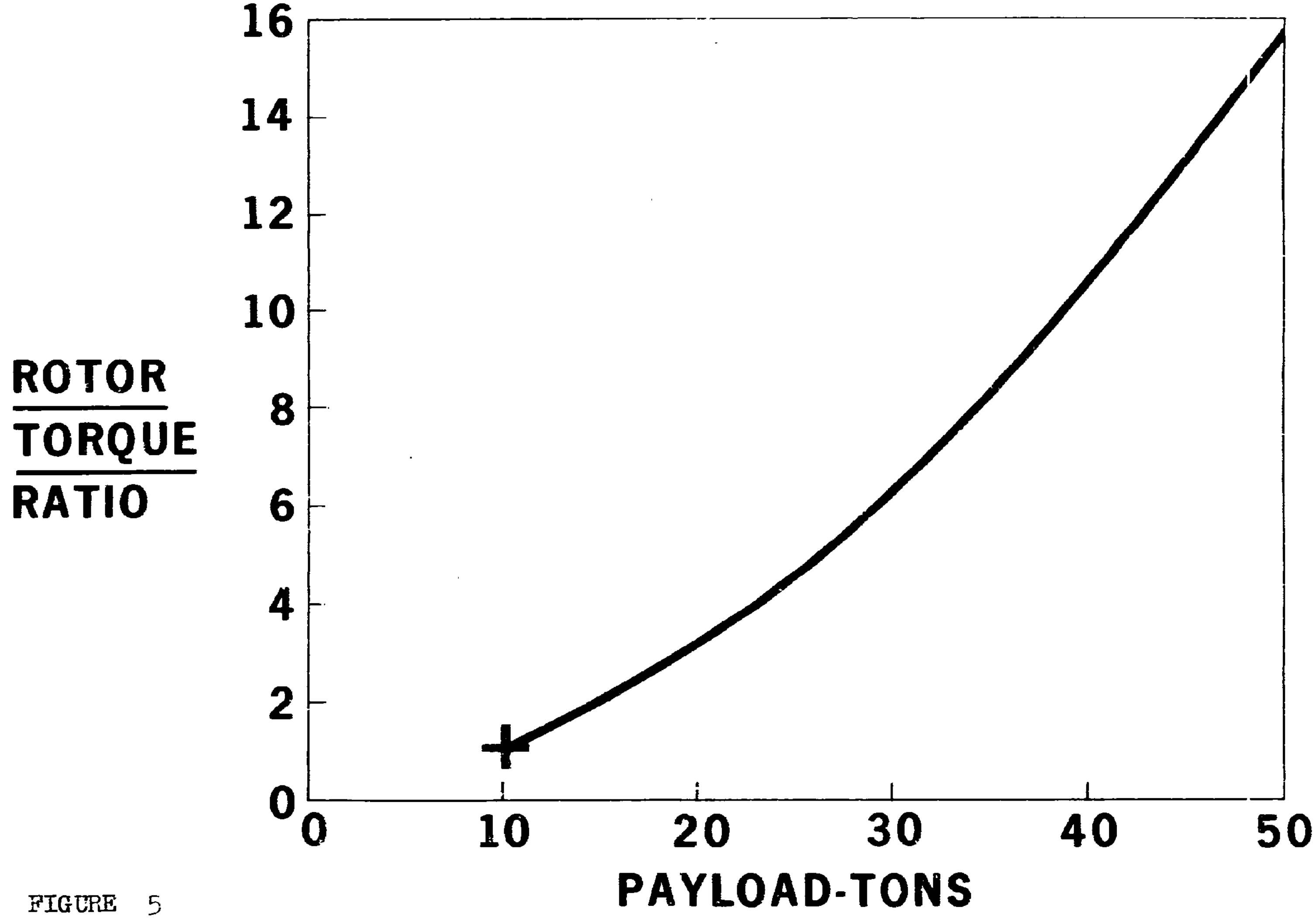


FIGURE 5

ROTOR DIAMETER VS FIRST FLIGHT DATE

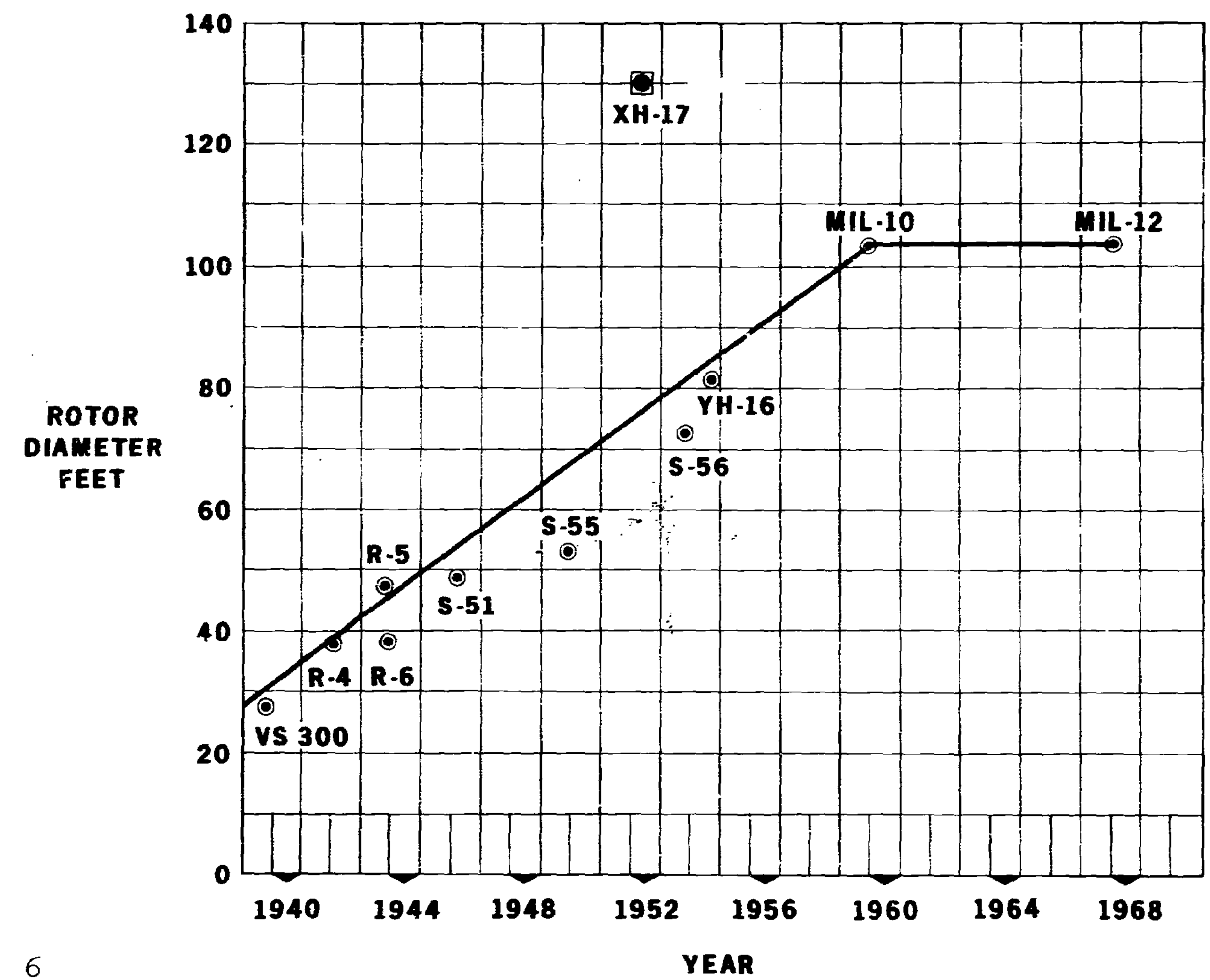


FIGURE 6

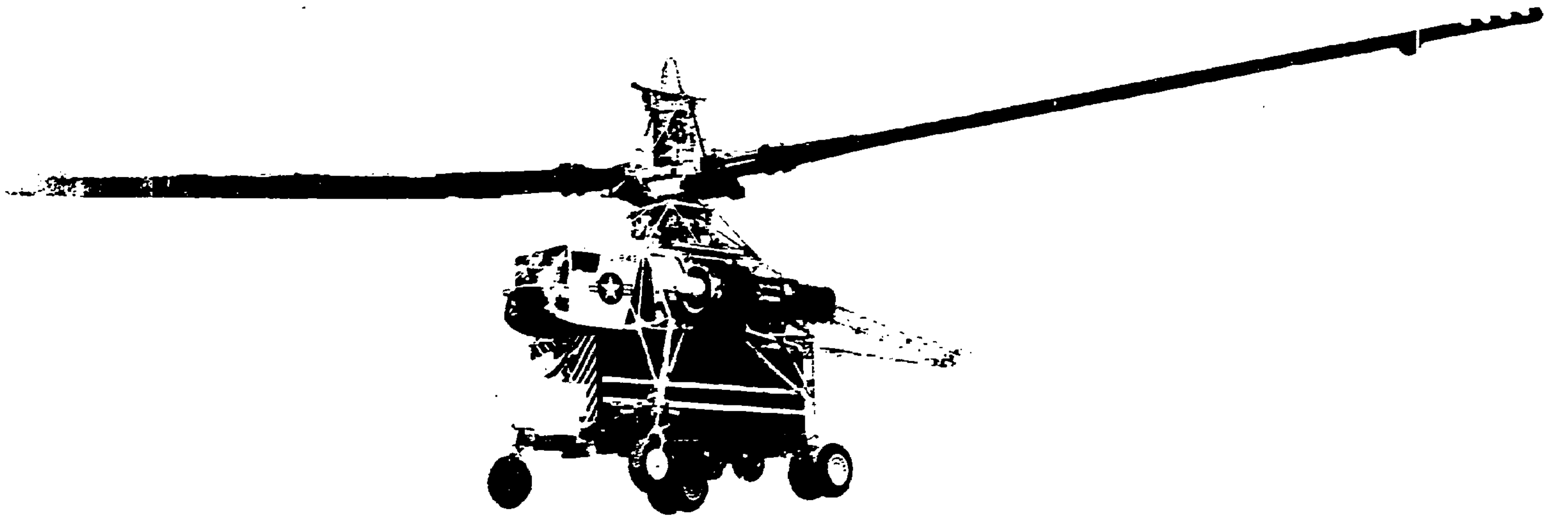


FIGURE 7

HUGHES XH-17 IN FLIGHT CARRYING LOAD

FIGURE 8

HUGHES XV-9A HOT CYCLE RESEARCH AIRCRAFT

HLH - EFFECT OF HOVER CRITERION ON USEFUL LOAD

HOVER CRITERION	4000' 95°F	S.L. 95°F	S.L. STD
EMPTY WEIGHT, LB	50,000	50,000*	50,000*
USEFUL LOAD, LB	77,000	97,000	111,000
GROSS WEIGHT, LB	127,000	147,000*	161,000*

*ASSUMING REDUCED LOAD FACTORS AND/OR USE OF SLINGS (NO HOIST) FOR LOAD HANDLING - NO ENGINE OR TRANSMISSION LIMITS ARE EXCEEDED

FIGURE 9

TIP PROPULSION TECHNOLOGY ADVANCES

CONFIGURATION	SFC, LB/RHP-HR	T_{GAS}, °F	V_J
RAMJET, PULSE JET	5 - 10	3200	1720
TIP BURNING PRESSURE JET	2 - 3	3000	2960
COLD CYCLE PRESSURE JET	1.5 - 3	360	1790
HOT CYCLE (XV-9)	~ 1.0	1200	2140
FANJET HOT CYCLE			
A) AVAILABLE ENGINE	0.85	825	1860
B) ADVANCED ENGINE	0.78	800	1980

FIGURE 10

SFC COMPARISON

SHAFT DRIVE VS FANJET DRIVE

	100% MILITARY ROTOR POWER		50% MILITARY ROTOR POWER	
	SHAFT DRIVE	FANJET DRIVE	SHAFT DRIVE	FANJET DRIVE
$SFC_{G\ HP}$	.357	.348	.408	.414
$SFC_{E\ HP}$	.43	-	.492	-
MRHP/G HP	.705	.424	.705	.483
$SFC_{MR\ HP}$	.506	.821	.579	.857
SFC RATIO	1.00	1.62	1.00	1.48

FIGURE 11

WEIGHT BREAKDOWN COMPARISON

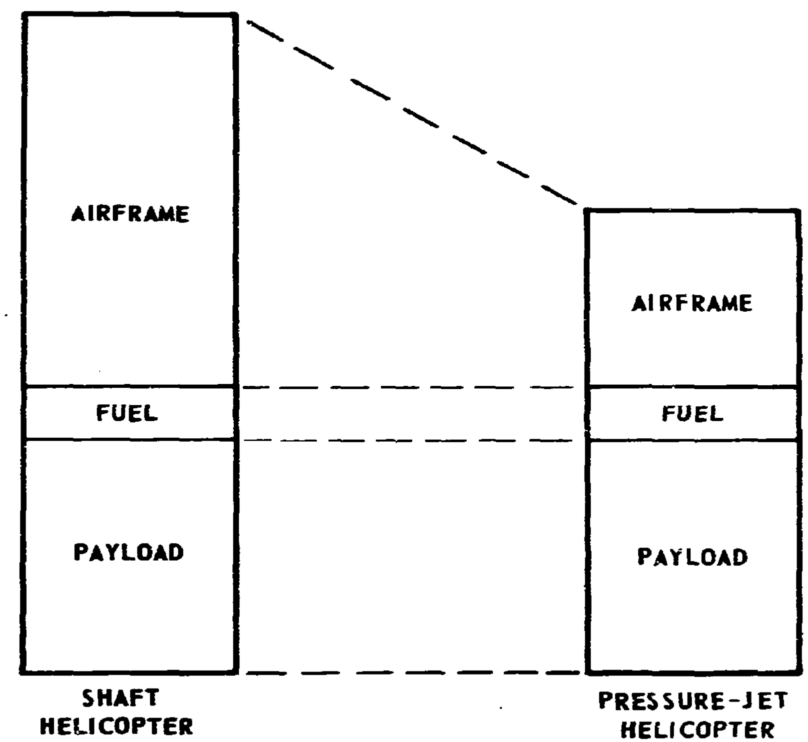


FIGURE 12

HOT CYCLE PROPULSION ANALYSIS

KEY PARAMETERS

- ENGINE TEMPERATURE AND PRESSURE
- PRESSURE DROP, ENGINE TO BLADE ROOT
- BLADE PRESSURE CHANGE

  DUCT FRICTION

  EFFECT OF CENTRIFUGAL FORCE
- NOZZLE VELOCITY COEFFICIENT

FIGURE 13

ROTOR SYSTEM TETHER TEST SETUP

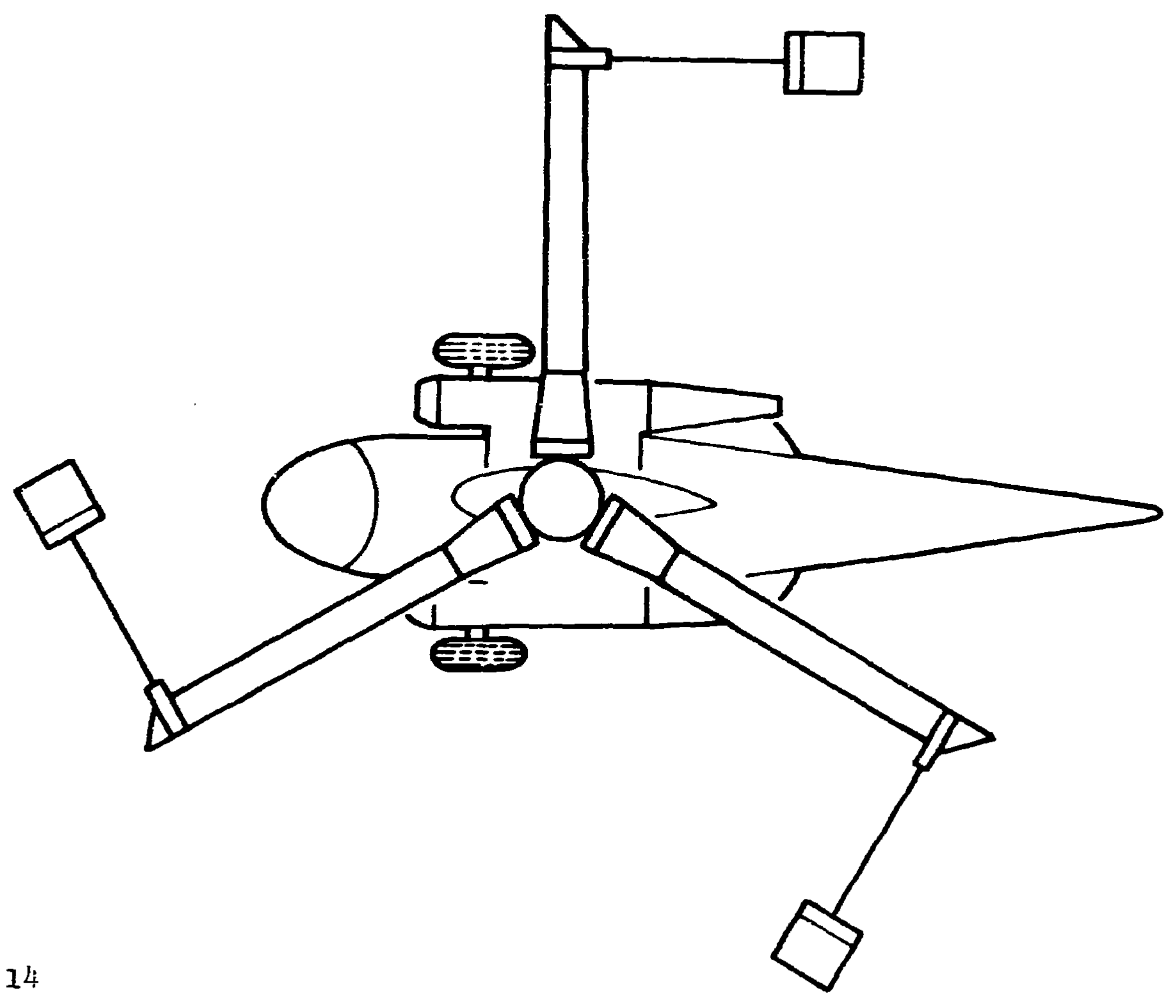


FIGURE 14

TETHER TEST BLADE DUCT FRICTION FACTOR DETERMINATION

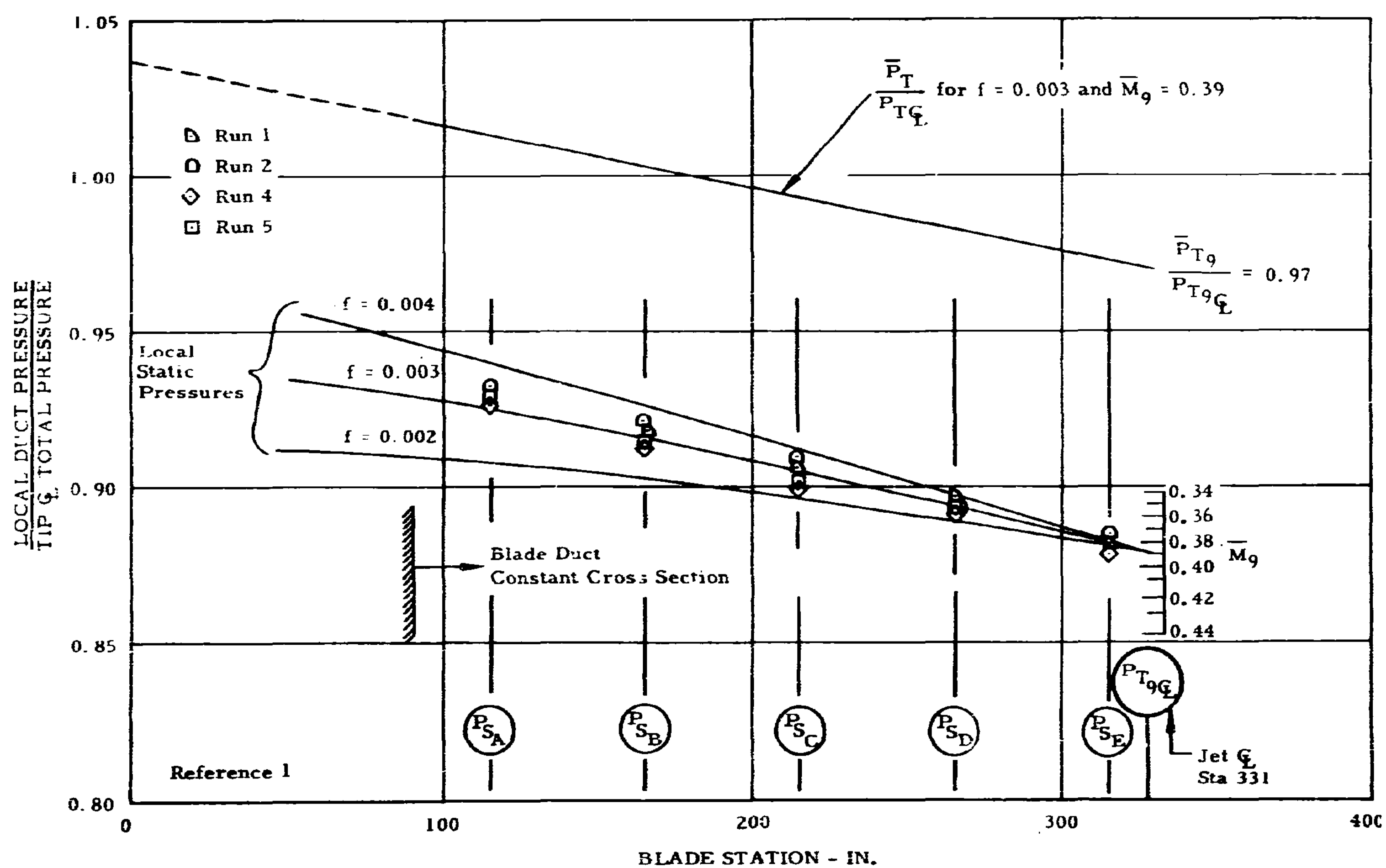


FIGURE 15

TETHER TEST VELOCITY COEFFICIENT

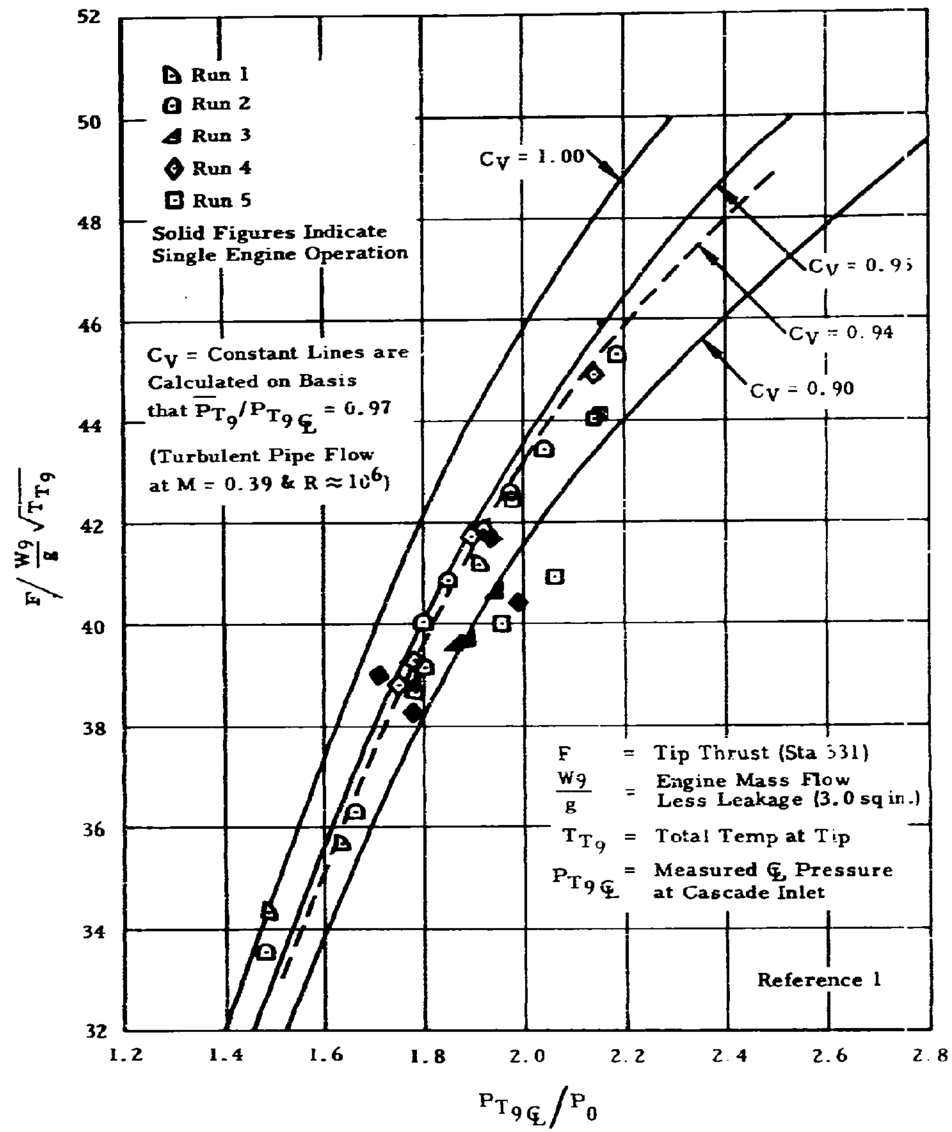


FIGURE 16

HLH ENGINES

	AVAILABLE	MODIFICATION OF AVAILABLE	DEVELOPMENT SCHEDULED
FANJET DRIVE	TF41 (IN PRODUCTION FOR A7D)	JTF10A-17 (TF 30 PROD. IMP. FOR F111D AIRPLANE)	JTF 22A-21 GE1/10F10 } COMPETING FOR F14B/F15 AIRPLANES
SHAFT DRIVE	NONE	J52 JET CONVERTED TO SHAFT ENGINE GE-1 JET CONVERTED TO SHAFT ENGINE	NONE

FIGURE 17

BLADE CROSS SECTION

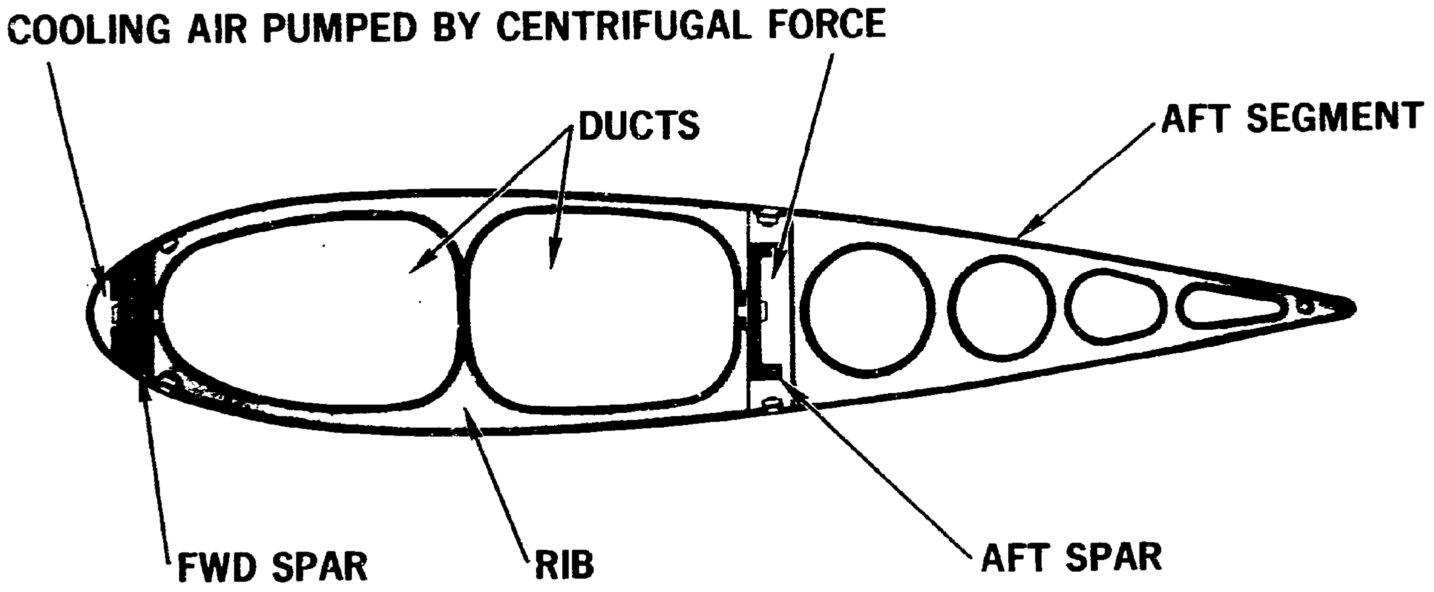


FIGURE 18

XV-9 BLADE TEMPERATURES

TIP STATION °F

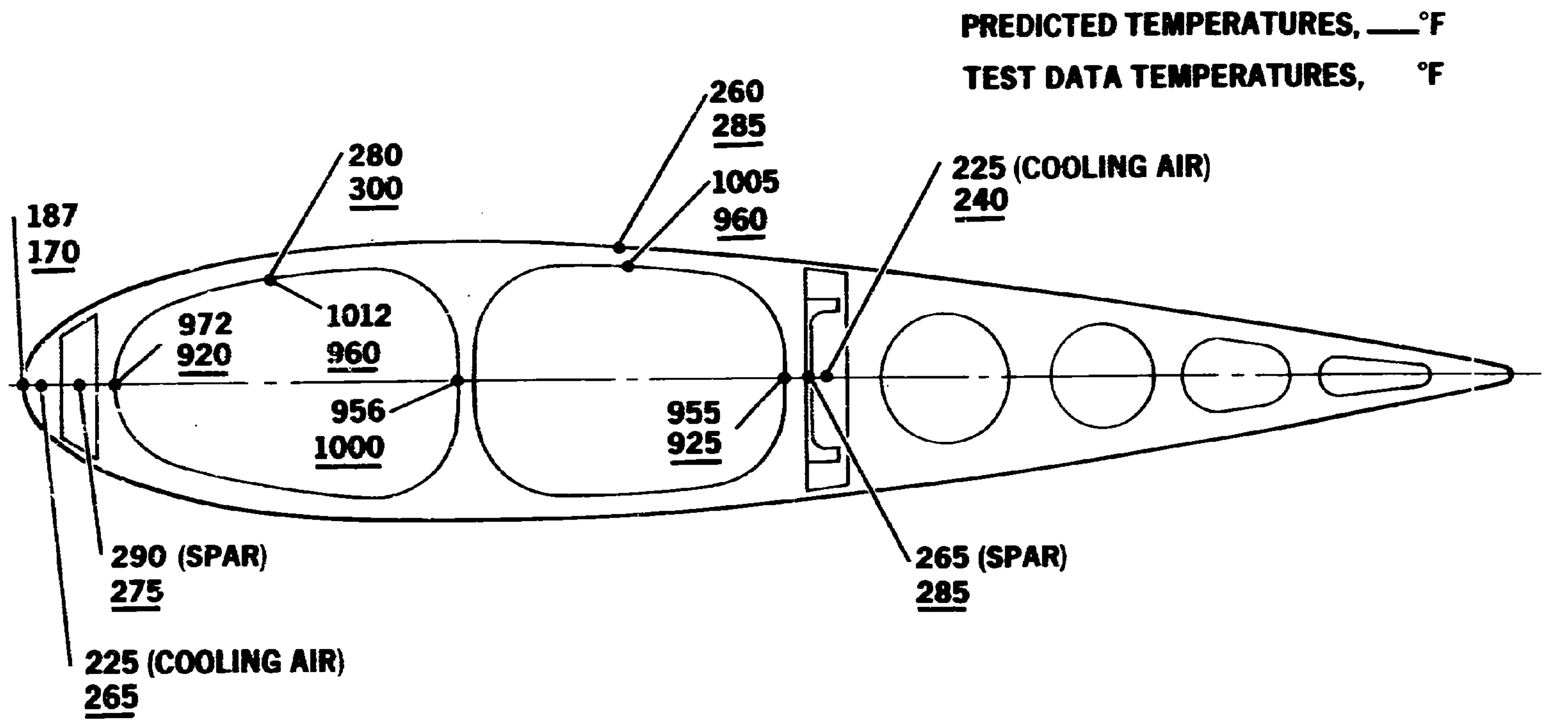


FIGURE 19

XV-9 DUCT SEALS

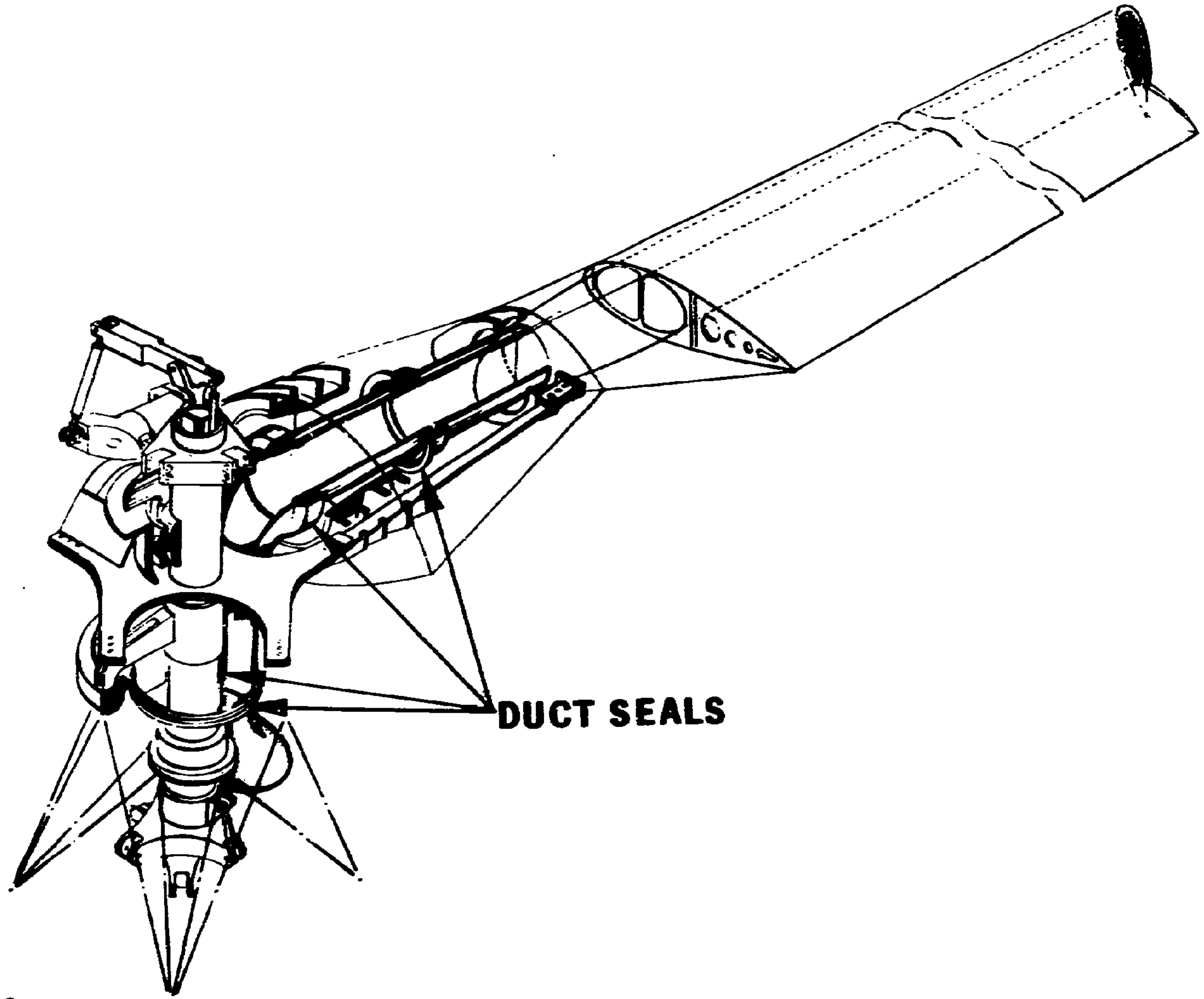


FIGURE 20

ROTATING SEAL OF XV-9A RESEARCH HELICOPTER

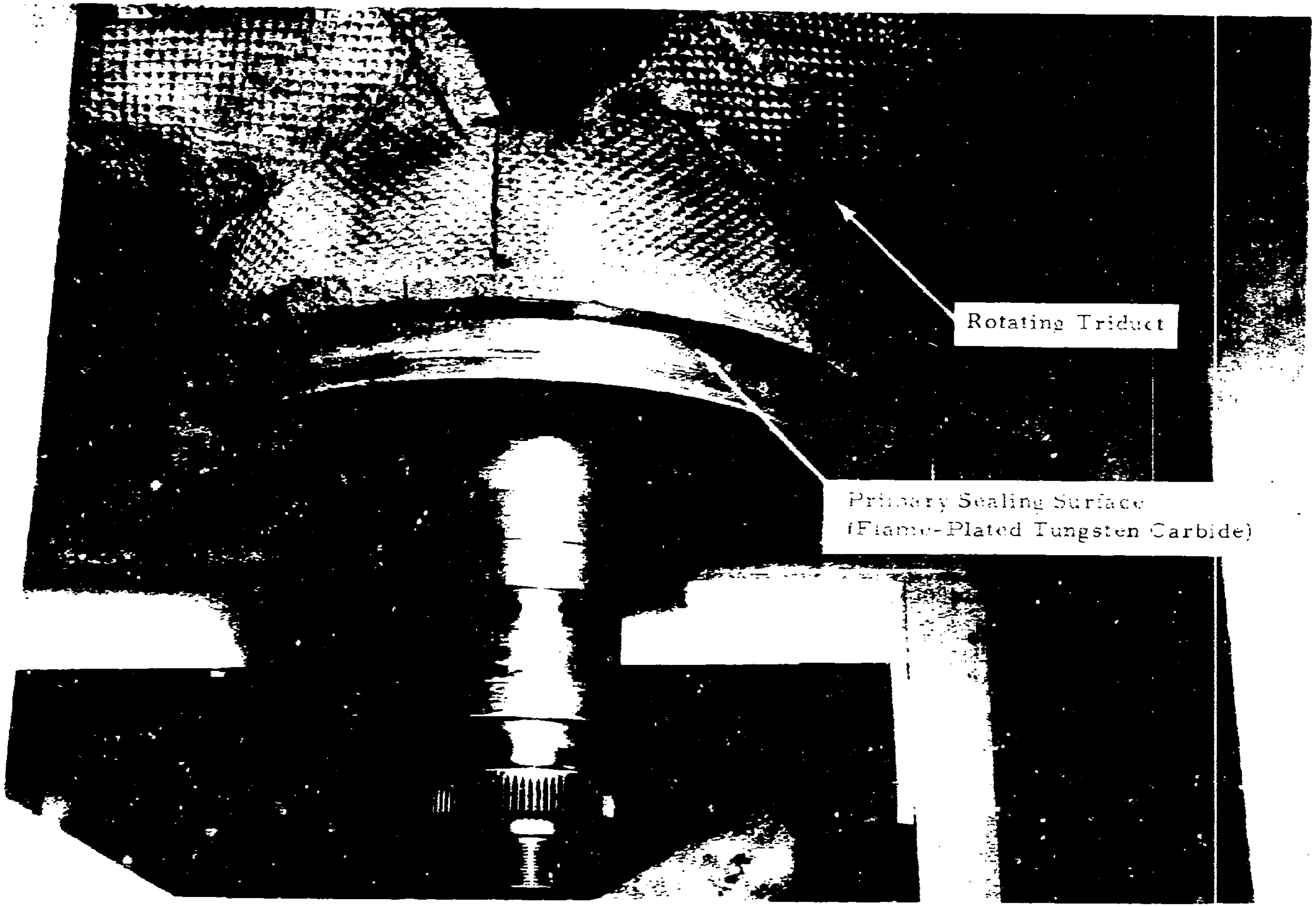


FIGURE 91

200-14

RESULTS FROM GROUND AND FLIGHT TESTS OF XV-9 HOT CYCLE RESEARCH AIRCRAFT

- 160 HOURS OF SATISFACTORY OPERATION
- PREDICTED HOVER PERFORMANCE VERIFIED
- UL/EW CAPABILITY $\approx$ 1.0 WITH HOVER CEILING OF 6000′ 95 °F BY USING PRODUCTION T-64 ENGINES AND MINOR PRODUCTION MODIFICATIONS
- NO TEMPERATURE PROBLEMS
- NEGLIGIBLE GAS LEAKAGE
- GOOD BLADE DYNAMIC CHARACTERISTICS
- ACCEPTABLE NOISE LEVEL
- LOW MAINTENANCE

FIGURE 22

COST EFFECTIVE COMPARISON OF HEAVY LIFT HELICOPTER SYSTEMS

Presentation by Mr. T. Schonlau - Hughes Tool Company

The Hughes Tool Company-Aircraft Division has conducted a cost effectiveness comparison of heavy lift helicopter systems (Figure 1). The main objective (Figure 2) of this study was to compare a fanjet-driven helicopter with a shaft-driven helicopter system. Sixty-three fanjet candidates and 63 shaft-drive candidates were utilized in this study. The conclusions of this study revealed that the fanjet HLHS was consistently more cost effective than the shaft-drive HLHS and that the 55- to 60-ton Hughes fanjet was the most capable and cost effective HLHS.

In this study the combat scenario (Figure 3) utilized, considered a helicopter availability of 20 helicopters which represents 75 percent of a 27 aircraft HLH battalion. Payload characteristics considered the movement of four representative Army units. Tactical payloads varying from 30 to 60 tons were considered. Personnel transported were within the fuselage and the number varied from 22 to 34 people. Two load carrying techniques were considered, a sling load for large vehicles and a gondola for composite loads. The gondola was basically 27 by 40 feet with a single point hoist attachment. It was composed of 6 pinned sections 9 feet by 20 feet, easily transportable on standard Army vehicles. For all the sorties considered, tactical integrity was maintained for single sorties loads. A one day combat supply was also included in each sortie load.

The mission profiles utilized for this study varied from 10 to 100 nautical miles. Two round trips were assumed without refueling.

During operations we assumed 15 minutes of hover time per round trip. Staging area and landing zone dead times were included in the mission time. Loads were available on demand and loads discharge space was available on demand. There was no queueing involved.

The tactical loading model (Figure 4) considered in this analysis included four representative tactical loads. First was the Armored or Mechanized Division Field Artillery Howitzer Battalion with 8" self-propelled and 155mm self-propelled equipment. The second tactical load was a Mechanized Infantry Battalion Armored or Mechanized Division. The third tactical load was a Light Brigade, consisting of Mechanized Infantry Battalion, Armored or Mechanized Division and a Tank Company, Armored, Infantry or Mechanized Division Tank Battalion. The final load was a Tank

Battalion, Armored, Infantry, or Mechanized Division. The last two loads considered required a 55-ton lift capability to handle the major equipment. These loads are representative loads for a tactical lift. The tactical lift was addressed in this study because when tactical and administration lifts are considered the tactical lift proves the limiting case.

The sorties versus payload for the self-propelled Field Artillery Howitzer Battalion is shown in Figure 5. This is a typical curve. We notice that step functions occur as the mission payload increases. That is, as we gain more capability we can cut down the number of sorties. For the lift of the self-propelled Field Artillery Battalion we see that the number of sorties required to complete the mission varies from 38 to 62.

The cost analysis (Figure 6) conducted during this study was based on CONUS Operations. Results obtained show that for a 200 unit buy the HLHS cost per flying hour for the 30-ton fanjet was $2,105. For a comparable shaft machine the cost was $2,920 per flight hour. The variation in cost is shown on the slide. As can be seen various elements contribute to the cost differential. Lower costs for RDT&E, Investment and Operation and Maintenance all contribute to the lower fanjet costs.

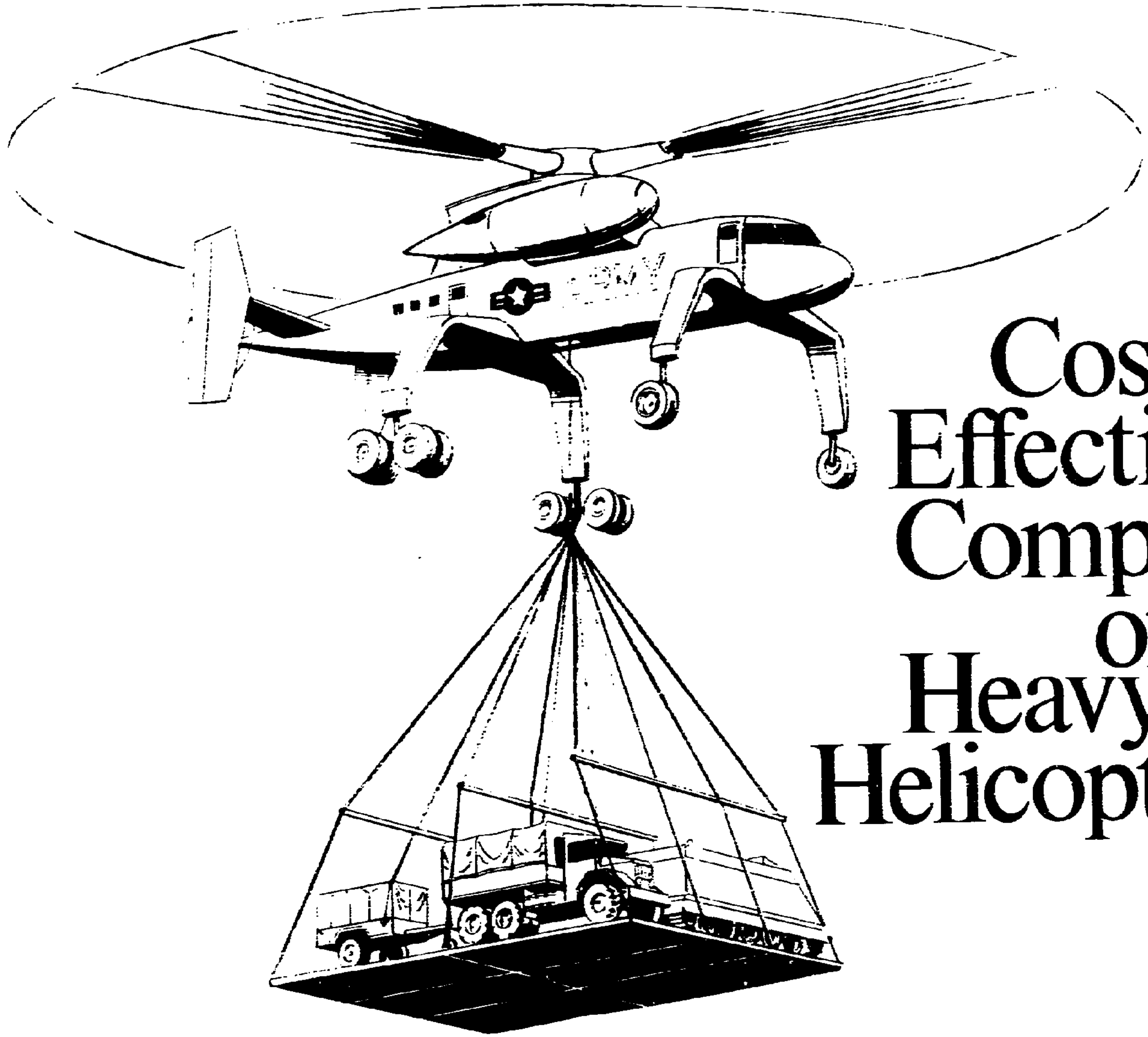
Systems cost for a 55-ton payload HLHS is shown in Figure 7. Assuming CONUS Operation with 900 flight hours per year and a buy of 200 units the cost per flight hour for the 55-ton fanjet was $3,410. For the 55-ton shaft-driven helicopter the cost per flight hour was $6,000.

The system effectiveness criteria utilized for this study was basically that developed by James Baines and based on utiles. Since this is simply an economic value to allow comparison a maximum utile value of 50 was selected. Using this approach allows us to evaluate the system effectiveness of the various HLHS candidate configurations. The curve shown on Figure 8 is derived from the equations shown on the lower portion of the figure for the indicated bounds. It was assumed that for any time less than 30 minutes mission time there would be complete confusion at the unloading area and thus the total utiles would be expended. For mission time greater than 10 hours there would be no value at all to the commander and, therefore, a utile value of zero exists. For times between 30-minutes and 10 hours a linear variation in utile values would exist. The cost effectiveness term Y_{ijk} is based on the i_{TH} candidate performing the j_{TH} mission over the k_{TH} radius.

The relative system worth utilized for this analysis is defined by the relative worth ratio shown on Figure 9. It is simply the system worth for the Hughes fanjet HLH divided by the system worth for the shaft-driven HLHS.

Results obtained from the study were presented as carpet plots to display relative system worth as a function of payload capability and mission range. A typical plot for the self propelled Field Artillery Howitzer Battalion tactical mission is shown on Figure 10. As can be seen, payload capabilities varied from 30 tons to 60 tons. Mission range varied from 10 miles to 100 miles. The relative system worth as shown reveals that the fanjet HLHS is 1.4 to 2.6 times as cost effective as an equivalent shaft machine. The carpet plot for the other missions have not been included due to time and space limitations. However, similar results were obtained for the Mechanized Infantry Battalion. For the light brigade and tank battalion the carpet plot compressed to relative system worth of from 2.0 to 2.5 for the fanjet HLHS.

The conclusions obtained from this study (Figure 11) show that the fanjet HLHS was 40 percent to 160 percent more cost effective that a comparable shaft-driven HLHS. Up to the maximum system worth design point the larger the payload the greater the systems worth. The study revealed that the shaft-drive HLHS has a maximum system worth for a payload of 35 tons. The fanjet HLHS has a maximum system worth for a 55-60-ton payload. It must also be pointed out that the 35-ton fanjet HLHS has a greater system worth than the 35-ton shaft-drive HLHS. The third point determined from this study revealed that the HLHS design payload requirements should be the maximum that technology permits at an acceptable risk level. The fourth and most important point was that the 50-60-ton payload HLHS has the greatest system worth.



Cost Effectiveness Comparison of Heavy Lift Helicopter Systems

FIGURE 1

HUGHES TOOL COMPANY / AIRCRAFT DIVISION / CULVER CITY, CALIFORNIA

SUMMARY

OBJECTIVE

- HLH SYSTEM COST-EFFECTIVENESS COMPARISON
  - FANJET HELICOPTER
  - SHAFT-DRIVEN HELICOPTER

CONCLUSIONS

- FANJET HLH(s) CONSISTENTLY MORE COST-EFFECTIVE THAN SHAFT-DRIVEN HLH(s)
- 55 TO 60-TON HUGHES FANJET MOST CAPABLE AND COST-EFFECTIVE HLHS

FIGURE 2

COMBAT SCENARIO

HELICOPTER OPERATIONAL AVAILABILITY

- 20 HELICOPTERS WHICH REPRESENT 75 PERCENT OF A 27-AIRCRAFT HLH BATTALION

PAYLOAD CHARACTERISTICS

- MOVEMENT OF FOUR REPRESENTATIVE ARMY UNITS
- TACTICAL PAYLOADS: 30-, 35-, 40-, 45-, 50-, 55-, AND 60-TON
- PERSONNEL TRANSPORTED WITHIN FUSELAGE FROM 22 TO 34
- LOAD-CARRYING TECHNIQUE
  - SLING LOADS: SOME VEHICLES
  - GONDOLA: VEHICLES, EQUIPMENT, AND BULK AMMUNITION (27-BY 40-FOOT SINGLE-POINT HOISTED)
- TACTICAL INTEGRITY MAINTAINED FOR SINGLE-SORTIE LOAD

MISSION PROFILE

- MISSION RADII: 10-, 25-, 40-, 50-, 60-, 75-, 90- AND 100-NAUTICAL MILE
- TWO ROUND TRIPS WITHOUT REFUELING

OPERATION

- 15 MINUTES OF HOVER TIME PER ROUND TRIP
- STAGING AREA AND LANDING ZONE DEAD TIMES INCLUDED IN MISSION TIME
- LOADS AVAILABLE UPON DEMAND
- LOADS DISCHARGE SPACE IS AVAILABLE ON DEMAND

FIGURE 3

TACTICAL LOADING MODEL

ESTABLISH TACTICAL LOADS FOR:

- ARMORED OR MECHANIZED DIVISION FIELD ARTILLERY HOWITZER BATTALION 155-MM, 8-IN., SELF-PROPELLED (TOE 6-355E)
- MECHANIZED INFANTRY BATTALION, ARMORED OR MECHANIZED DIVISION (TOE 7-45E)
- LIGHT BRIGADE, CONSISTING OF
  - MECHANIZED INFANTRY BATTALION, ARMORED OR MECHANIZED DIVISION (TOE 7-45E)
  - TANK COMPANY, ARMORED, INFANTRY OR MECHANIZED DIVISION TANK BATTALION (TOE 17-37E)
- TANK BATTALION, ARMORED, INFANTRY OR MECHANIZED DIVISION (TOE 17-35E)

FIGURE 4

SORTIES VERSUS PAYLOAD

SP FIELD ARTILLERY HOWITZER BATTALION (TOE 6-355E)

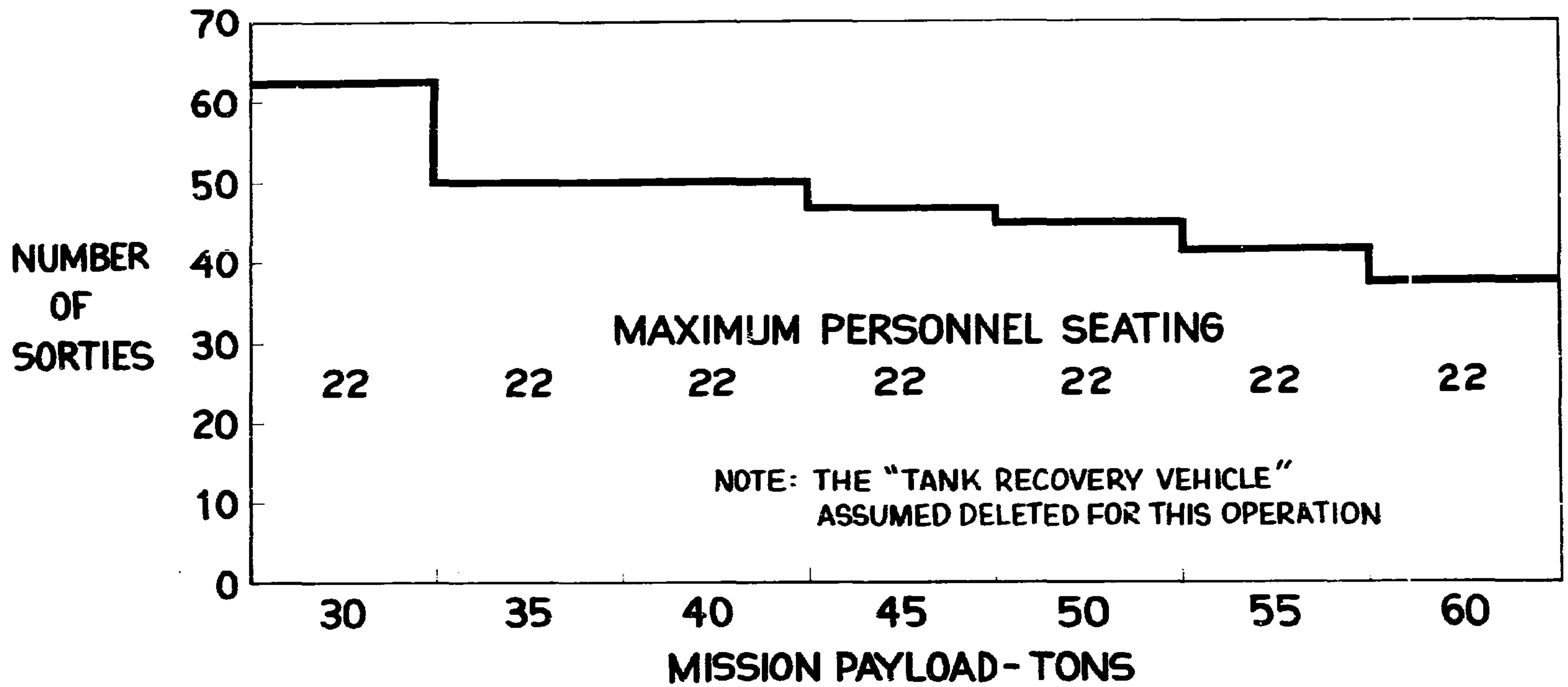


FIGURE 5

30 TON PAYLOAD HLH 10-YEAR SYSTEMS COST CONUS OPERATION

($ X 10^6 BASED ON A BUY OF 200 UNITS)

(900 FLT-HR/YR)

COST CATEGORY	FANJET HELICOPTER	PERCENT	SHAFT-DRIVEN HELICOPTER	PERCENT
I RDT & E	0.98	5.17	1.31	5.00
II INVESTMENT	5.58	29.41	9.81	37.35
A. FLYAWAY	4.30	22.65	7.55	28.74
B. MAINTENANCE FLOAT AND INITIAL TRAINING	1.07	5.65	1.89	7.20
C. GSE	0.21	1.11	0.37	1.41
III O & M	12.40	65.42	15.14	57.65
DEPOT MAINTENANCE	1.44	7.60	2.50	9.52
ORGANIZATIONAL MAINTENANCE	2.59	13.59	4.45	16.92
DIRECT OPERATING COST (FUEL)	1.65	8.71	1.23	4.68
CREW PAY, TRAINING SUPPORT	6.72	35.43	6.96	26.53
TOTAL 10-YR SYSTEMS COST PER HLH	18.96	100	26.26	100
HLH SYSTEMS COST $/FLT-HR	2105		2920	

FIGURE 6

55 TON PAYLOAD HLH 10-YEAR SYSTEMS COST CONUS OPERATION

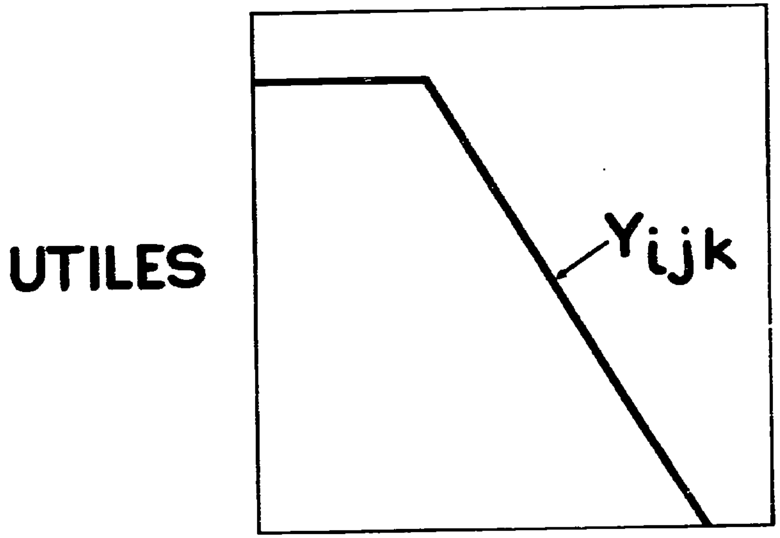
($\$ \times 10^6$ BASED ON A BUY OF 200 UNITS)

(900 FLT-HR/YR)

COST CATEGORY	FANJET HELICOPTER	PERCENT	SHAFT-DRIVEN HELICOPTER	PERCENT
I RDT & E	1.93	6.29	2.81	5.20
II INVESTMENT	9.25	30.12	18.36	34.00
A. FLYAWAY	6.91	22.50	14.60	27.06
B. MAINTENANCE FLOAT AND INITIAL TRAINING	1.95	6.35	3.13	5.78
C. GSE	0.39	1.27	0.63	1.16
III O & M	19.51	63.59	32.84	60.80
DEPOT MAINTENANCE	3.42	11.13	7.00	12.94
ORGANIZATIONAL MAINTENANCE	5.13	16.70	10.70	19.79
DIRECT OPERATING COST (FUEL)	2.80	9.12	2.60	4.81
CREW PAY, TRAINING, SUPPORT	8.16	26.64	12.54	23.26
TOTAL 10-YR SYSTEMS COST PER HLH	30.69	100	54.01	100
HLH SYSTEMS COST $/FLT-HR	3410		6,000	

FIGURE 7

SYSTEM EFFECTIVENESS



MISSION EFFECTIVENESS
(MISSION TIME INTERVAL HOURS)

$Y_{ijk} = 5.2631\ T_{ijk} - 52.631 \qquad 0.5 < T_{ijk} < 10.0$

$Y_{ijk} = 50 \qquad T_{ijk} < 0.5$

$Y_{ijk} = 0 \qquad T_{ijk} > 10.0$

FIGURE 8

RELATIVE SYSTEM WORTH

$$\text{RELATIVE WORTH RATIO} = \frac{\text{SYSTEM WORTH HUGHES FANJET HLH}}{\text{SYSTEM WORTH SHAFT DRIVEN HLH}}$$

FIGURE 9

RELATIVE SYSTEMS WORTH

SELF PROPELLED FIELD ARTILLERY HOWITZER BATTALION

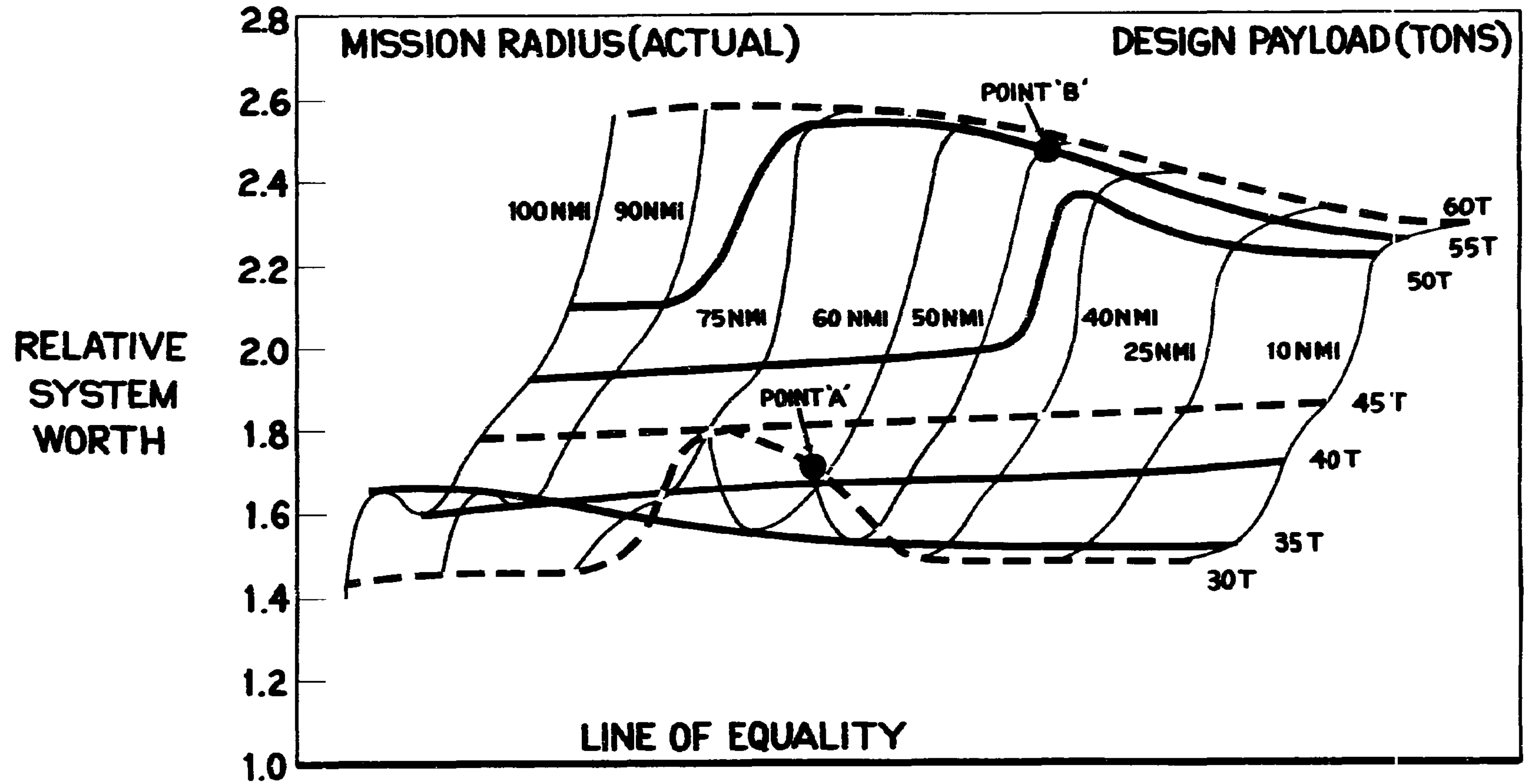


FIGURE 10

CONCLUSIONS

- FANJET IS 40% TO 160% MORE COST-EFFECTIVE THAN EQUIVALENT SHAFT-DRIVEN HLHS
- UP TO MAX SYSTEM WORTH DESIGN POINT THE LARGER THE P/L THE GREATER THE SYSTEM WORTH

  SHAFT-DRIVE MAX SYSTEM WORTH PT = 35 TON P/L
  FANJET MAX SYSTEM WORTH PT = 55-60 TON P/L
- THE DESIGN P/L REQUIREMENT SHOULD BE THE MAX THAT TECHNOLOGY PERMITS AT AN ACCEPTABLE RISK LEVEL
- 55 TO 60 TON P/L HLHS HAS GREATEST TACTICAL SYSTEM WORTH

FIGURE 11

APPENDIX K
LIGHTER-THAN-AIR VEHICLE TECHNOLOGY

Presentation by Dr. R. Ross-Goodyear Aerospace Corporation

Much of the work on LTA or, at least, most of the presentations-are by the real LTA enthusiasts - we sometimes call "helium heads". They sort of believe that one can do anything with LTA. If somebody wanted a Mach 2.0 aircraft, they'd find a way to do it with an airship. Even if they'd have to get to the moon they'd find a way. Most of the time it is kind of emotional. There is a wonderous sentimental attachment to lighter than air, and because of that you find that real enthusiasts just want to do everything with it. That was what happened when silicone came out; people wanted to do everything with it from brushing your teeth to using it for lubricants. Some places it works very well; other places it doesn't. It is kind of interesting to note though that if you actually look at the technology and forget about the emotions and study applications, there are places where you can do things with lighter than air vehicles that you can't do with anything else as well. However, there are a lot of emotions tied up with the decisions as to whether you go this way or not.

It was kind of interesting to hear Dr. Burton's comments about the Russians and what is happening in their country. It sounds very much like what is happening here. One finds that the technical groups that are studying the subject are coming up with the fact that here is something that ought to be looked at, and then you will find people that say "I don't care how it comes out on paper, we don't want to be involved". It is interesting to see the similarity of attitudes toward LTA in the two countries.

At the present time I will try to tell you a little bit about lighter than air and we'll see if we can find where it fits into this picture. Really, when you set the ground rules, that decides what you are going to come out with in the answer. To give you an example: I have a friend who is a very good photographer. He used to go out and take pictures and he would go to the same place everybody else would, but when he came back his pictures were different somehow or other, and I asked him, "How come?" He said, "Well, when I go down with this group on the tour and they are pointing out some beautiful edifice and everybody is taking a picture of it, I turn around to see what they are missing. This is where I get my pictures."

Very often we get so much involved in what we are doing that we don't look elsewhere and maybe we are missing something very important. I think that this is one of the reasons why we were asked into this meeting. We are all very much involved with helicopters and they are doing a wonderful job and you can't beat them at their own game so maybe we should look around - look in the other direction and see if there isn't something there that we should be taking a picture of at the same time.

Now there are a lot of falacies that are built into us and one of them is the fact that helium is in low supply. At the time when rigid airships were important in this country and in Germany there was a very short supply of helium; we did not give helium to the Germans. As a matter of fact, I recall reading in the history books that there was only enough helium to fill one airship at a time and they had to decide whether they were going to fly this airship or that airship and put the helium in the ship that they were going to fly. They flew the Hindenburg and Graf Zeppelin with hydrogen. Now ever since the dramatic burning of the Hindenburg at Lakehurst, we celebrate it every year by reburning it on television. We all have seen these were most spectacular pictures. If you ask any of the kids - under thirty is a kid isn't it? - that is all they know. "Airships burn beautifully. Every year I watch them burn on television." That is the only thing they know about airships. Now as far as helium is concerned, I attended a helium symposium here in Washington a few months ago and I learned something that I didn't realize. I knew that it was possible to go out and buy helium, but I didn't realize that there are ten commercial companies in this country making or separating helium today and there is so much helium available that they are running around trying to find uses for it. I think they put 13 billion cubic feet back into the ground every year because they can't find uses. So there is lots of helium available.

It was interesting to note that at this very same symposium where they told about how much helium they have, there was another paper on the same morning where somebody got up and said that they were normally using 1.6 million cubic feet of helium. They had a new program going showing where that yearly amount could be cut into one-third because they were changing over to hydrogen slowly. So here within our own country, our own people are trying to save helium and we've got more helium than we know what to do with. I guess the statement they made there is that in their wildest dreams they can't see where we can possibly use up the helium we have today in this country in the next 100 years if they don't find any additional supply and they think they are going to find additional supplies. So helium is here if we want to use it. We just have to know that it is.

Now it is interesting to note that if we are going to try and decide technologically what is going to happen here we could come out with one answer but the statisticians often times look at the subject and they come out with their answer. I noticed that some curves were presented the other day showing what aircraft have done so far and an extrapolation of the curve. It is going to be pretty hard not to fall on that curve if we do a good job on future aircraft. You are probably going to have a pretty tough job to even get to the curve but you are probably not going to fall very far from it.

Just for the fun of it, we decided to look at lighter than air other than technologically. Since the beginning of the century it has gone through five cycles of interest and if we aren't going to spoil history, lighter than air is going to be here again. We are ready for the next cycle. That is nontechnical.

Well, now we'll get back to the technical side. The original vertical take-off aircraft was a balloon and I guess if we had a picture we could show that that is the way they got off the ground originally the first time. Then people started trying to move at will and they put power on it and we started powered balloons and worked our way up from there. Well, it depends on where you stand; what you see.

Figure 1 shows the relative use of airspace by various types of aircraft. Now, I am not saying that airships couldn't operate somewhere else. As a matter of fact we just finished a powered balloon program for the Air Force, and we were working somewhere in the vicinity of 60,000 feet. This is not an airship - just a powered balloon. So you can get and do a job there, but if you want to do a job where it will be at its best, you should be down here close to the ground. It doesn't mean that you can't fly at 4,000 feet at 95 degrees or any other requirement. This, however, is where you are going to get your best advantages - down close to the ground, since you sacrifice lift as you go on up.

Now I am going to give you a 30 second education or course on airship design, so after you leave here you will all be experts (Figure 2). We all know about the rigid airships, the Akron, Macon, the Hindenburg and the Graf Zeppelin, that type had a rigid frame and gas cells inside this frame to perform the lift function. Most of the airships that have been built in this country over the last 30 years or so have been the non-rigid or pressure stabilized type. They are really not non-rigid, they are very rigid. But they are one balloon that contains the gas and are held rigid by the fact that this gas is under pressure.

When we say "under pressure", we should define this a little better. I guess since the days when we were kids and played with balloons we knew that if you take a balloon and you stretch it and you touch it with a pin making a tiny hole, it bursts. There are many people who still think airships are like toy balloons and will burst if you make a little hole in them. The truth of the matter is that back in the days of the Civil War when people used to fly balloons made out of oiled paper they knew that this wasn't true. When a balloonist would take off and somebody would say "gee you've got a lot of holes in the side of your balloon" he would say "shucks, I guess I will have to come down early. I guess I won't be able to stay up all night." These people didn't know what an expandable balloon was so they didn't worry about them bursting. But since we know of expandable balloons as kids we think of airships in the same light. The truth of the matter is they are not expandable balloons. They are a very definite shape and the pressure inside is very, very low so that if you get a hole the gas leakage out of the balloon is very, very slow. Now, this comes back to the old point of vulnerability. Can you knock it down with a single shot? You find that you have to fire a lot of shots. As a matter of fact they had a balloon flying in Vietnam not long ago and it had some classified equipment on it. It started to get away and they sent up three helicopter gun ships and emptied everything they had in it. They couldn't get the thing to come down. So, as far as vulnerability is concerned this is an entirely new approach. Now, I am not saying you can't put a big missile up there and blow the thing apart because you certainly can. But it takes an awful big hole to make any kind of an effect on it. If you get a hole in the envelope you won't know for a long, long time if you even got hit. Our small airships fly back and forth across the country all the time. Periodically they are inspected to look for holes because somehow or other they are a nice attractive target and people like to shoot at them. The only time the pilot knows it is when it comes through the car. He doesn't particularly like that. Anyway we have essentially an envelope which is made out of a very strong material.

At one time the Achilles heel was the means of attaching the car, which is a big weight, to the envelope. It is done by means of distributing the loads into the surface through catenary-like arrangements. This attachment was the weak point at one time and we thought we'd never build any of these envelopes bigger than one million cubic feet. We have found means now whereby you can make an attachment here that can take any load that the material can take. The last airships built for the Navy were a million and a half cubic feet. At the present time we see no practical limit for this type of ship.

Now this airship does have inside it a number of aircells and these are there primarily so that as you change your altitude or you have temperature change or anything like that you don't have to waste your helium. In other words, the helium changes in volume and all you have to do is let some air in and out to compensate for it. Now the size of aircells is strictly a function of how high you want to go and if you are going to operate close to the ground, these can be relatively small. It just means that you can carry more helium and you can get more lift. That is all there is to it.

The car on the ship is very much like an airplane. The only thing is that you usually don't go above 20,000 feet so you usually don't get involved in pressurization. Now it isn't because you can't. It is because the airship's major roles are usually at low altitude. The car is typical aircraft construction; all the instruments, controls, engines, and everything else are aircraft-type. The tail surfaces are just ultra lightweight, small airplane-type design. I once was trying to give somebody a physical idea of size of a tail surface and compared it to the size of a room in a house. I found out I could put the whole house there, not only one room. We'll get an idea of the size a little later.

Figure 3 will give you a little feel of relative size. Here's a 707, here's a 747, and a C-5A. The ZPG-3W is the last blimp that we made for the Navy and the top one is the Akron. When people say something is large to us it seems very small. You could fly the 707 right through the Akron without touching the wings against the walls, if you had a pilot that good. Now this gives you the feel of size. When you tell somebody who is working on the Saturn that a Fourth of July rocket is big, he doesn't quite feel the same way about it. This was built in the late '20s and the early '30s. It was 40 years ago that they did this kind of thing so what could we do now? So, don't let size bother you at all. This is the Navy blimp around 400 and some feet long. We were working on that when the B-70 mockup was made. I went out to see the mockup and we all walked around it and everybody was oh-ing and ah-ing at the size. I had just come out of our air dock where we were working on this one and I thought that the B-70 looked like an awfully small ship.

Let's go on to Figure 4. As far as payloads are concerned, also, the numbers are usually quite a bit bigger. So when you are talking about getting into 50-ton or 60-ton payloads, it sounds like a real nice low value and we should be able to do it very quickly. This depicted a conventional airship. Usually you think of airships as being long-range because they use so little fuel. You see the change in payload is not very large. If you are talking about 25 miles or 50 miles there and back, you're here at the left of the chart. If you want for some reason or other to extend the range, it doesn't change the payload too much because they don't require very much power.

Incidentally, those sizes were all pretty much within what has already been made.

Now if you wanted to raise your cruise speed to 120 knots. (Figure 5), all that you are doing is increasing the power required which means that the curves become more steep. In other words, you start getting more of an effect due to the fact that you have to start to use up fuel, but you can see the category of what you want to do in payload is not at all out of line.

Now the secret of your application is to go back to the original idea and make a lighter-than-air vehicle, a VTOL (Figure 6). There is no reason in the world why you shouldn't be able to rise vertically, fly horizontally to where you want to go and land vertically. Once you start doing that, the idea came up that you can actually change the shape so that you get some dynamic lift out of it. The Navy has done a considerable amount of work in showing that airships, even conventional shapes, are dynamic lift vehicles. It wasn't at all out of line to put 10,000 pounds extra on a million and a half cubic foot airship and make a running take-off just like a heavier-than-air pilot does - pull back on the wheel, establish an angle of attack and climb. This was standard procedure. As a matter of fact the last ships had a tricycle landing gear and reversible propellers and when you came into land, you landed on your gear, put your propellers in reverse and came to a stop. You will notice that by doing this the Navy also improved by a considerable amount, the development of ground handling. In the old days of rigid airships, they flew in a balanced manner and when they came in and landed you saw hundreds of people around who helped to hang onto the ship and hold it still. The last ships that the Navy operated in the early '60s would come in and land and take-off again without any ground crew at all. The only time that they used a ground crew was when they were going to take the airship into its hangar or put it on its mast. At that time they used two tractors with, I think, two or three men on each tractor and this is about all you saw in the field. You had a mast and you had two tractors and they did all the ground handling with this equipment, so you can forget about this idea of lots of people holding them down because the ships were flown heavy and they essentially sat on the ground. They used a very short take-off. They were a STOL-type aircraft, a short run and they'd take-off. Now, if these vehicles had a vertical lifting propeller or rotor, you could visualize how they could lift themselves off vertically and then convert and fly horizontally. The idea of increasing the dynamic lift was applied not too long ago in balloons for the first time. It was probably the first time something new has happened in balloons for the last 50 years. This was done with a tethered balloon and was actually applied to logging operations in the western part of the country. What we did was to put two balloons together in the form of a V and add a tail on them so that they were stable and we had a dynamic lifting surface.

We found out that a normal, single-hull balloon that could lift about a ton statically, would give 200 pounds extra in dynamic lift. By taking this configuration and giving it some shape, operating it about 30 miles an hour, we were able to get not 10% in dynamic lift but 500%. In other words, this balloon that would normally lift a ton would now lift five tons. You can readily visualize if you are going to pick up logs and take them out of the forest, you would change the payload from 2,200 pounds to 10,000 pounds. If you examine the economics of it, at the end of the day it is how many pounds of logs you have at the landing that determines your effectiveness. So, the dynamic lift becomes a very important factor.

By shaping the airship a little differently you can now start getting a larger dynamic lift, but what does it do to the ground handling? If this vehicle is flying heavy then when you come in and land you are not subject to the vagaries of the wind. You are going to set where you come down. The heavier you make it the less apt you are to move so if you take-off vertically you can do that by the function of the amount of power you have. The gas then actually supports the bare structure and you might say that you are going to carry the payload with your dynamic lift. If you considered this a helicopter, all you are saying is that the helicopter doesn't weigh anything. You are balancing off the empty weight and you are just carrying your payloads with the rotor so when you look at the horsepower required it is, naturally, considerably less. The percent of heaviness you carry, 25% or 80% is up to you. What we were showing is that there is no need for heavy-duty runways, or expensive airports. This vehicle is very much like the helicopter. Actually the buoyancy of the gas is used to support the structure so you have a structure with zero weight. It is hard to beat that.

This ship has a new name (Figure 7). We call this a "DYNASTAT" instead of airship. In some places the word airship doesn't go over too well, so we take an Aerostat and mix it with dynamic lift and come up with a "DYNASTAT". Now, everybody can look at that because it's new and feasible. You can select the lift that a "DYNASTAT" should have and determine the size it would be. You remember in the conventional airship, these lines were fairly flat, but now you are starting to get some of this square cube law and you begin to see a little bit of an effect of range. When you talk about a low range you can see that you get substantial lift. It doesn't take a lot of speed to do it either.

Now, let's talk about safety. Visualize the "DYNASTAT" with some percent heaviness and say that you had a complete power failure. If you are already in forward motion, you can glide to a flared landing. If you are hovering, you are in the parachute range, as far as wing loading is concerned. Just look at the vehicle - the size of it, and the amount of overweight and you find that you have a loading that is very similar to parachutes so you are actually descending at a parachute rate. In other words, you aren't catastrophic. It is almost impossible to be catastrophic in this kind of a vehicle.

Now, may I tell you where we stand today (Figure 8)? We are looking at a small, silent airship. There are a lot of applications, as I mentioned earlier. There are people who like to count wildlife from the air. We were told that in one place in Africa, where they were trying to keep track of the elephants, they almost wiped out a herd because they were counting the same elephants over and over. They thought that this herd was growing like mad and they told the hunters to go in and shoot them only to find that they were the same elephants they were counting all the time and they almost wiped them out. So, they would like to have some kind of a quiet vehicle that would travel overhead without disturbing this wildlife.

You find also that whenever you talk about coming in close to a city or coming in close to people you want a quiet vehicle. We have an airship out advertising all the time and sometimes we think that the noise attracts the people who looked at it. The other day, I understand that this backfired. An airship flew over a concert that was going on- an open-air concert, and the conductor didn't appreciate the engine noise. So we need a quiet one even for our advertising or we are going to get adverse effects from it. We are also doing some work for the military services on a little program we call "Joe II", looking at an airship that could do a particular job that they had in mind.

We have been flying these little advertising airships and, incidentally, to give you a feel of size, they are about one-tenth the size of the last ones that we built for the Navy. They are only about 150,000 cubic feet in volume against one and half million. They are the ones that you see coming overhead all the time.

As far as lighter-than-air is concerned, the company finds that is their best advertising dollar. No matter what you do, fly airplanes overhead, put movies out, blare noises and all this, people somehow or another like to see airships and when we put one overhead they all look up at it. We have a big flying billboard. Right now we are making three more airships-they just finished one and they've got two more under construction. There are three ships in our hangar at the present time. The three new advertising airships flying around the country with fantastic signs on the side are actually big billboards with four-colored lights in motion and the cost of the sign alone is probably a million dollars.

The company has found that this medium has wonderful operating experience. They have been running these airships this same size, essentially for almost 40 years without any fatalities at all. I am not saying that they did not have some accidents. As a matter of fact, I was recently in Indianapolis where we had taken General Wassel up for a ride in the blimp. That night the fellows were supposed to bring the blimp back to Akron. I don't know what happened but it was two o'clock in the morning when they ran into some high tension lines that were 80 feet off the ground. They must have been following a highway and I guess these lines went across their path and they ran right into them. Well, the airship got tangled up in the lines. They lowered a rope and climbed down to the ground. We sent our chief pilot back out there and with a cherry picker put him back into the car. By dropping weights and valving gas and juggling, he was able to get the blimp loose from the wires. He flew over to a little field where they did some emergency repairs. They then flew it home and worked like mad to meet a commitment at the Toronto Exposition in Canada a week later. But, they have been operating for years and years and years and records show they are probably one of the safest vehicles you could make.

Now there is work going on toward the development of a quiet engine. I just picked out the Lear work here on steam engines as an example, and if you want to get into some quiet operations this is one way of making essentially a silent vehicle. I think this is one of the points which is going to come up later on.

As far as the transports are concerned, we are now talking about something bigger - something that might have some dynamic lift. We made some studies for NASA and NASA made some studies for themselves, and it becomes apparent that this is a good way to carry a heavy payload. We made studies for the Air Force and they showed that it was a good way to do their job too. Many companies in the United States have contacted us with problems of moving specific items. The studies always show that an airship transporter is a feasible solution.

I have shown here a nuclear transporter. There is no reason why today you couldn't make a nuclear powered lighter-than-air vehicle and this is not at all out of line. Many years ago we acutally proposed that we put a blimp in the air that could be nuclear powered. We could have had a nuclear power plant flying around to prove the reliability, endurance and everything else that was questioned. We have been working with the Air Force showing how this same nuclear power plant could be used also in a stretched version of the C-5A, for example, for a first nuclear airplane. In case of the airship, however, you would end up with a vehicle that would not only have an infinite endurance but would also have a tremendous payload. We have been working closely with Westinghouse and the NASA people. All of the nuclear powered engine components would only take a reasonable development period. No breakthroughs or anything like that are needed. Let us go on to the next one.

Let me give you a feel of the time schedule associated with these aircraft (Figure 9). Back in the early years here was the Akron, Macon and Hindenberg. The left scale indicates their size at that time. Shown are all the different blimps that we did for the Navy and our advertising ships. Also, here is a silent one that we are now working on. Here are what we called a transporter or an atomic powered unit. So you see, we are not really talking about unbelievably big vehicles. We are talking about things that are within the realm of reality. The people who built the early airships had to have guts because with the little bit of knowledge that they had about weather and structures they came out with some fantastic vehicles. If you took today's knowledge and applied it to this chart, it would be no sweat to come up with a decent vehicle. Let me give you a feel of what these are. Let us take one of these transporters, say a five million cubic foot ship (Figure 10). We would be able to have a top speed of maybe 140 knots on that with 160 thousand pound payload. Well, maybe you would take 68 thousand pounds of it as dynamic lift with a useful lift of 160 thousand.

Figure 11 is not necessarily how it would look. The artist had to have some license but I thought this would be representative. We can make point to point delivery. Incidentally, when looking at this vehicle as a transporter, we found that almost any company has a parking lot big enough to operate in and out of. We may have to take down a couple of lamp posts, but you would be able to come into their parking lot, load, and leave.

Reduce cargo vibration or cargo shock. I don't know if you realize it or not but they have never recorded a half "g" in an airship. The highest I think was .427 and that was done by making almost a crash landing where the envelope almost enveloped the car it came down so hard. You don't have seat belts or anything like that in an airship. It is an extremely gentle ride. It is almost impossible to move an item on the ground with that low a loading. Today we spend lots of money making special containers to move fancy equipment because we have no other way of doing it. An airship just normally carries it that way.

All-weather operation. I don't know if you realize it or not but the Navy actually conducted a test in cooperation with NASA and they went out looking for icing conditions that they could not fly in. They ran this test, I believe, for four years and they went all over the country looking for some bad conditions. Whenever they found one, their idea was to penetrate to the point where they would have trouble, turn around and come out. They had this ship equipped with ice-measuring equipment and detectors of all types. They never, in that entire period, found something they could not handle. They also conducted some tests under normal operating conditions. They planned on operating a mission somewhere off the East Coast. When the day came they went out to conduct this mission, to remain on station for a ten day period. They happened to encounter the worst weather in 75 years when it occurred. All other aircraft transportation was grounded but they kept on operating. So, I think that they have been able to prove that lighter-than-air vehicles can be the most all-weather aircraft that are flying today.

Now, this does not say you can't have problems. The worst thing that you have in an airship is freezing rain. When you are flying along and encounter freezing rain, it piles up on the large surface. However, freezing rain is present only in a very, very limited area and all you have to do is get out of it; know that you are getting into it, change your altitude and you are out of the freezing rain. Regular icing does not occur or snow does not accumulate on the envelope. It does, however, collect on antennas or any other little thin protuberance. The big contours, it just goes right over. Well, I think those are the main things shown here except that you certainly do have VTOL. Ice can accumulate though quite rapidly, depending on the meteorological conditions, but not too fast for you to get out of it. You see, especially with the dynamic lift shape, you can carry tremendous loads. On conventional airships they never had a problem as far as I know. If they are able to detect it, they climb up out of it or descend to get out of it. To the best of my knowledge I don't know of any icing instances they could not handle.

Figure 12 shows that if you went to a ten million cubic foot vehicle at the same speed, you might have a useful lift here of 275,000 pounds.

Now, let us look at a big one just for a second (Figure 13). I know that this is larger than what you were thinking of but we are talking in the vicinity of 800,000 or 1,000,000 pound payload and you could make it either a chemical or nuclear vehicle. If you wish, the speeds may go up to about 200 miles an hour in the nuclear version. You might say "why that fast?" Really if you look at the drag curve it goes up quite rapidly but we find that if you put a nuclear engine in this ship, this speed is possible. It is interesting to note that the Russians came up with about the same size and power. This is because you can end up with about the same weight for shield and power plant for something that is about 20,000 horsepower or 60,000 horsepower and so you might as well put the 60,000 horsepower unit in there which just means that you can then go on up to this speed if you want to. If you looked at a curve of power vs. weight, it would show that a certain basic weight is needed for any power setting and the weight does not start to increase until a substantial amount of power is available. So if you go to this much weight you might as well go to that point in the curve where it starts to increase and use the power available. This has been looked at for military applications.

The next chart (Figure 14) shows a commercial application of a large transporter. I just put this one in here because a company did approach us on moving houses and established the sizes they might be. They have factories going now where they build houses but economically they can only deliver the houses to the places immediately around the factory. The further they can go from the factory, the more area they can cover with these houses. This is a square function. If they had a vehicle that could pick the house up at the factory and carry it to where they want it and put it down, they would be able to make the economics look good. Their weights go all the way up to a million pounds per house. They want to build the entire house in the factory and to pick the whole thing up. They even want to pour the concrete in the most inexpensive way. Incidentally, as far as speeds are concerned, if you find that you are going to travel 25 miles, there is no point in going Mach 2.0. If you include the take-off and the landing time you might as well go at a low velocity. The only time you'd need the high velocity would be for going great distances.

If you want to look at a time schedule (Figure 15), I don't think we have to go all the way to the '90s to do it, but I should point out that this is the technical time scale. In other words, we could actually build these small ones right now and deliver them as we are doing, in a year.

If you go to this intermediate one, and if you started it right now it would probably be the end of '72 before we would have one flying. If you looked at the giant one, we could probably build the first one by the end of '73, the nuclear one, probably not until the end of '74. It would be at least one year longer to get the nuclear engine. But I am showing you that this does not require a breakthrough. It is just normal development. However, this does not include the tough part of the job and that is getting the signature on the piece of paper that tells you to go, or getting the approvals afterwards. This is just the engineering work involved, and if they don't start this until 1990, we can't build it in 1974. This, at least, gives you a feel. Let us look at the next one real quickly.

I took this one time (Figure 16) and broke it down a little further. Just to define the ship that you want and what you could do with it would take a year to do. You actually have to do some testing, not a tremendous amount - just to more or less optimize some of the things that you are getting into. The chemical ship can be built in three years, the nuclear one in a four-year period of time. Now that is as far as the technology is concerned. This does not involve the decision as to which service would operate it or anything else.

UTILIZATION OF AIRSPACE

ALTITUDE-FT. ↑	
70,000	SST
40,000 – 30,000	JET TRANSPORTS
30,000 – 10,000	PRIVATE CORPORATE AIRCRAFT
10,000 – 3,000	PRIVATE, SMALL AIRCRAFT, HELICOPTERS
3,000 – 0	LTA

FIGURE 1

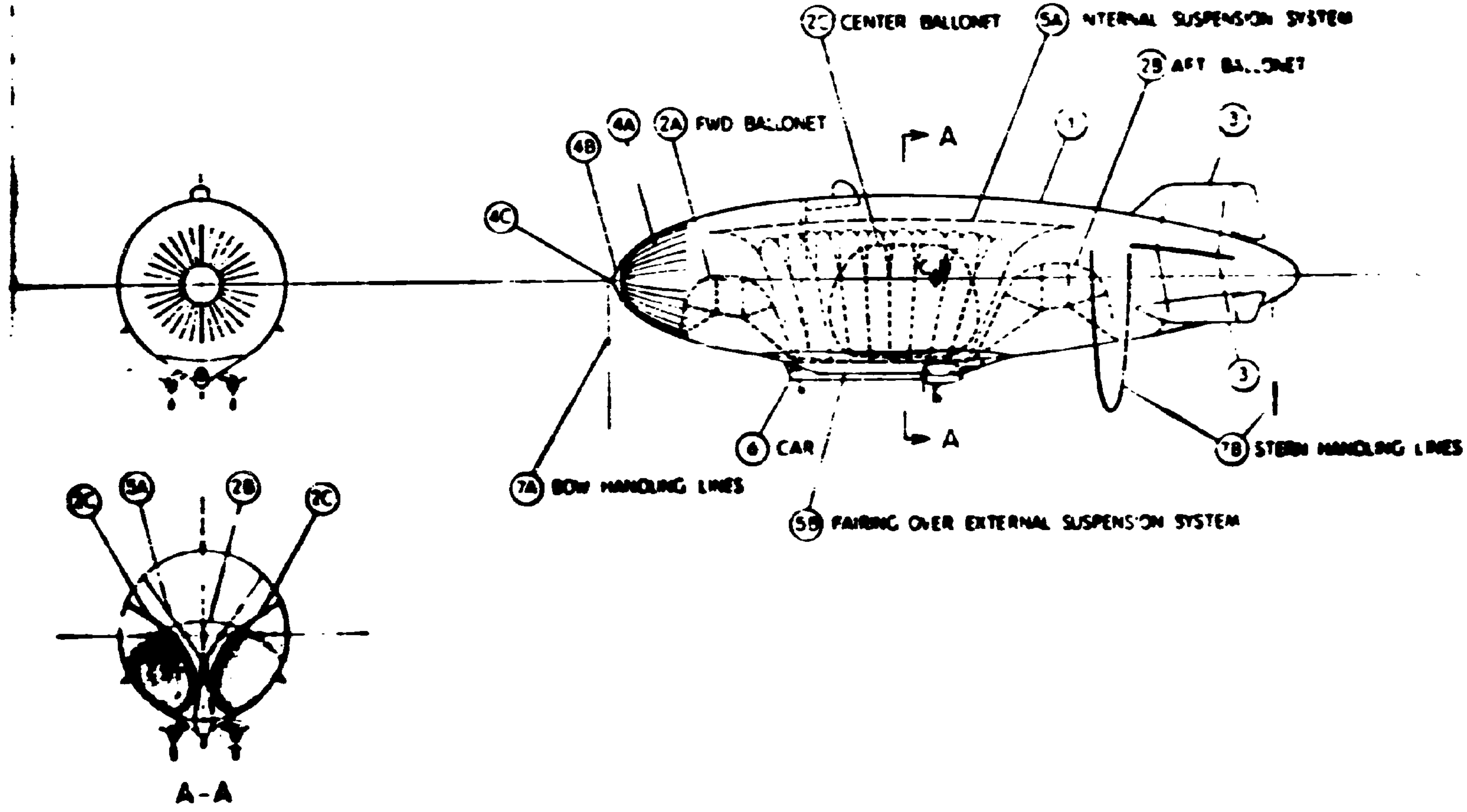


FIGURE 2

200-14

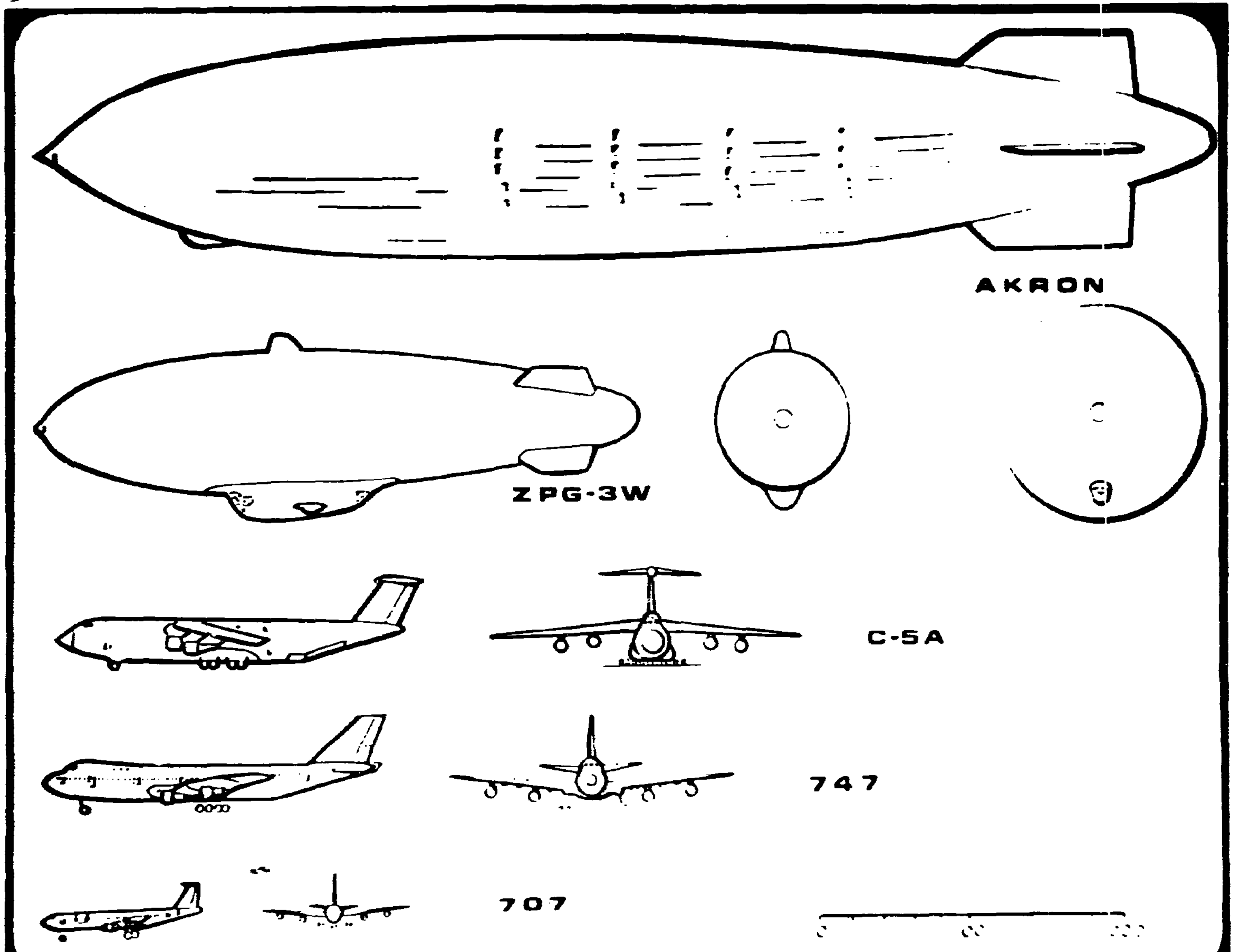


FIGURE 3

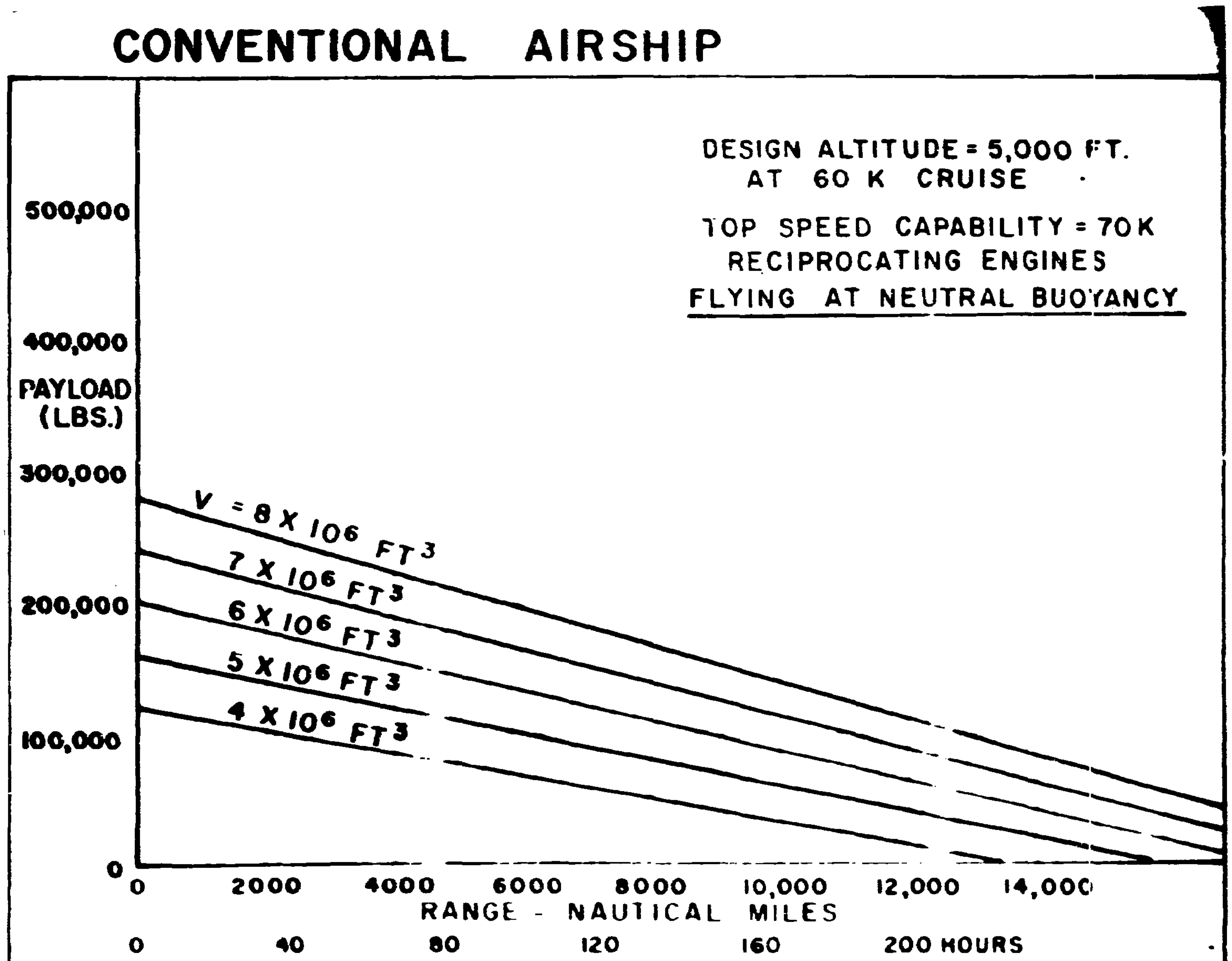
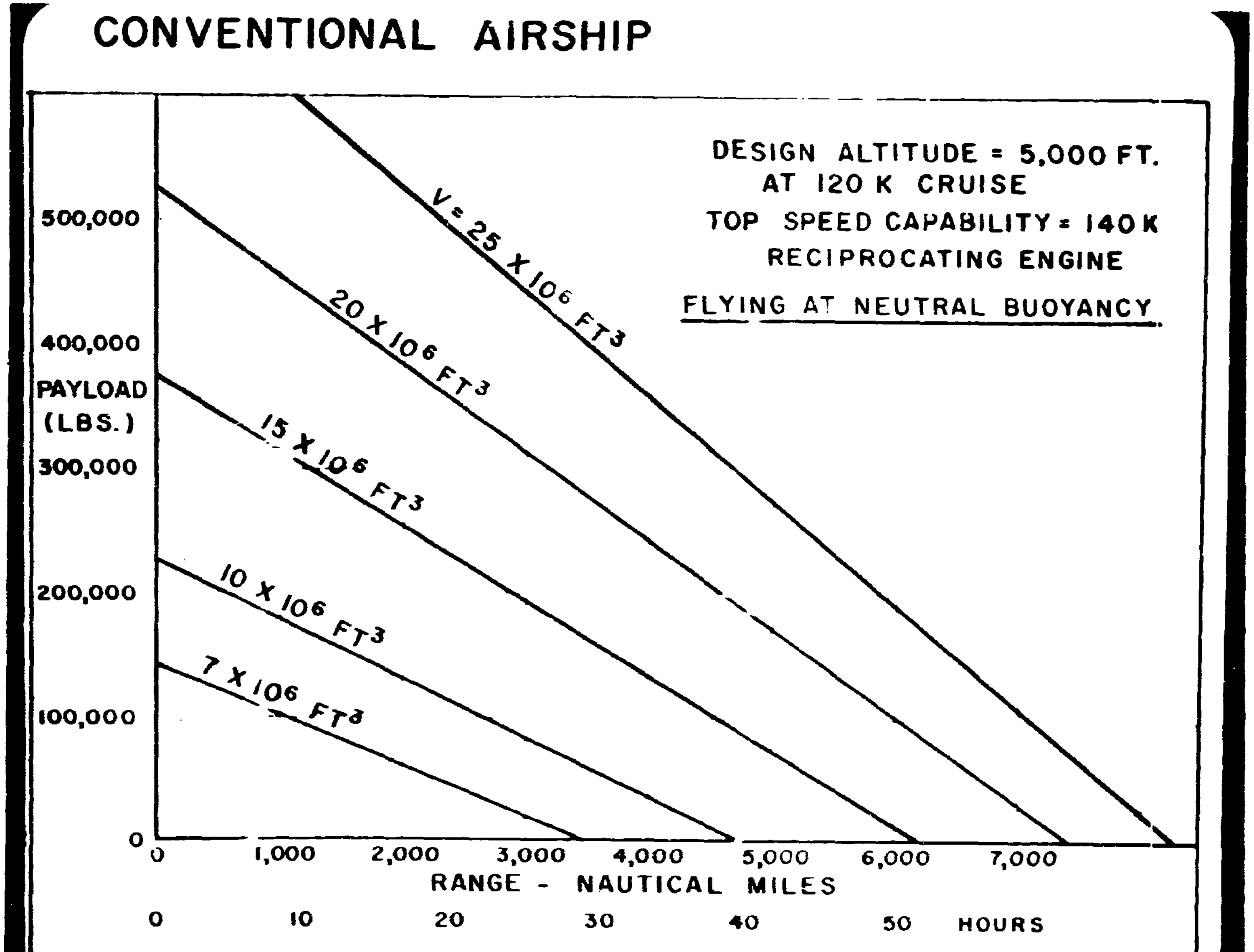

CONVENTIONAL AIRSHIP
DESIGN ALTITUDE = 5,000 FT.
AT 60 K CRUISE
TOP SPEED CAPABILITY = 70 K
RECIPROCATING ENGINES
FLYING AT NEUTRAL BUOYANCY
500,000
400,000
PAYLOAD (LBS.)
300,000
200,000
100,000
0
V = 8 X 10^6 FT^3
7 X 10^6 FT^3
6 X 10^6 FT^3
5 X 10^6 FT^3
4 X 10^6 FT^3
0
2000
4000
6000
8000
10,000
12,000
14,000
RANGE - NAUTICAL MILES
0
40
80
120
160
200 HOURS

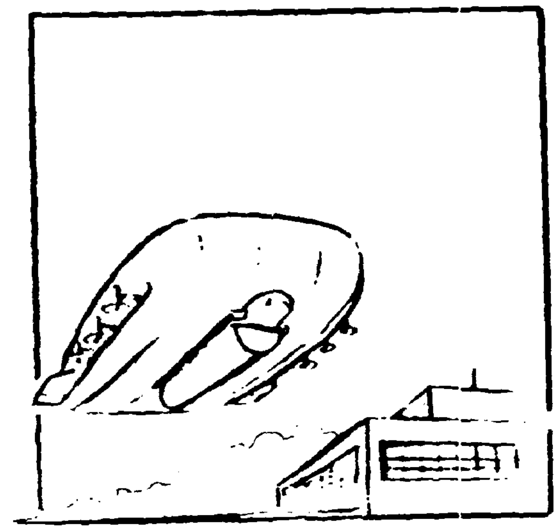
FIGURE 4



CONVENTIONAL AIRSHIP
DESIGN ALTITUDE = 5,000 FT.
AT 120 K CRUISE
TOP SPEED CAPABILITY = 140 K
RECIPROCATING ENGINE
FLYING AT NEUTRAL BUOYANCY
V = 25 X 106 FT3
20 X 106 FT3
15 X 106 FT3
10 X 106 FT3
7 X 106 FT3
500,000
400,000
PAYLOAD
(LBS.)
300,000
200,000
100,000
0
0
1,000
2,000
3,000
4,000
5,000
6,000
7,000
RANGE - NAUTICAL MILES
0
10
20
30
40
50
HOURS

FIGURE 5

V.T.O.L.



-NO HEAVY DUTY RUNWAYS ARE REQUIRED

-NO EXPENSIVE AIRPORTS ARE NEEDED

-NO VALUABLE LAND HAS TO BE APPROPRIATED TO HANDLE AIRSHIPS

BECAUSE THE MODERN AIRSHIP DERIVES LIFTING POWER FROM BUOYANCY IN ADDITION TO DYNAMICS OF FLIGHT. IT CAN LAND PRACTICALLY AT THE DOOR OF A FACTORY AND AT THE FOOT OF A GANTRY, OPERATING INTO AND FROM FIELDS TOO SMALL FOR HEAVIER-THAN-AIR AIRCRAFT OF COMPARABLE CARGO CAPABILITY.

FIGURE 6

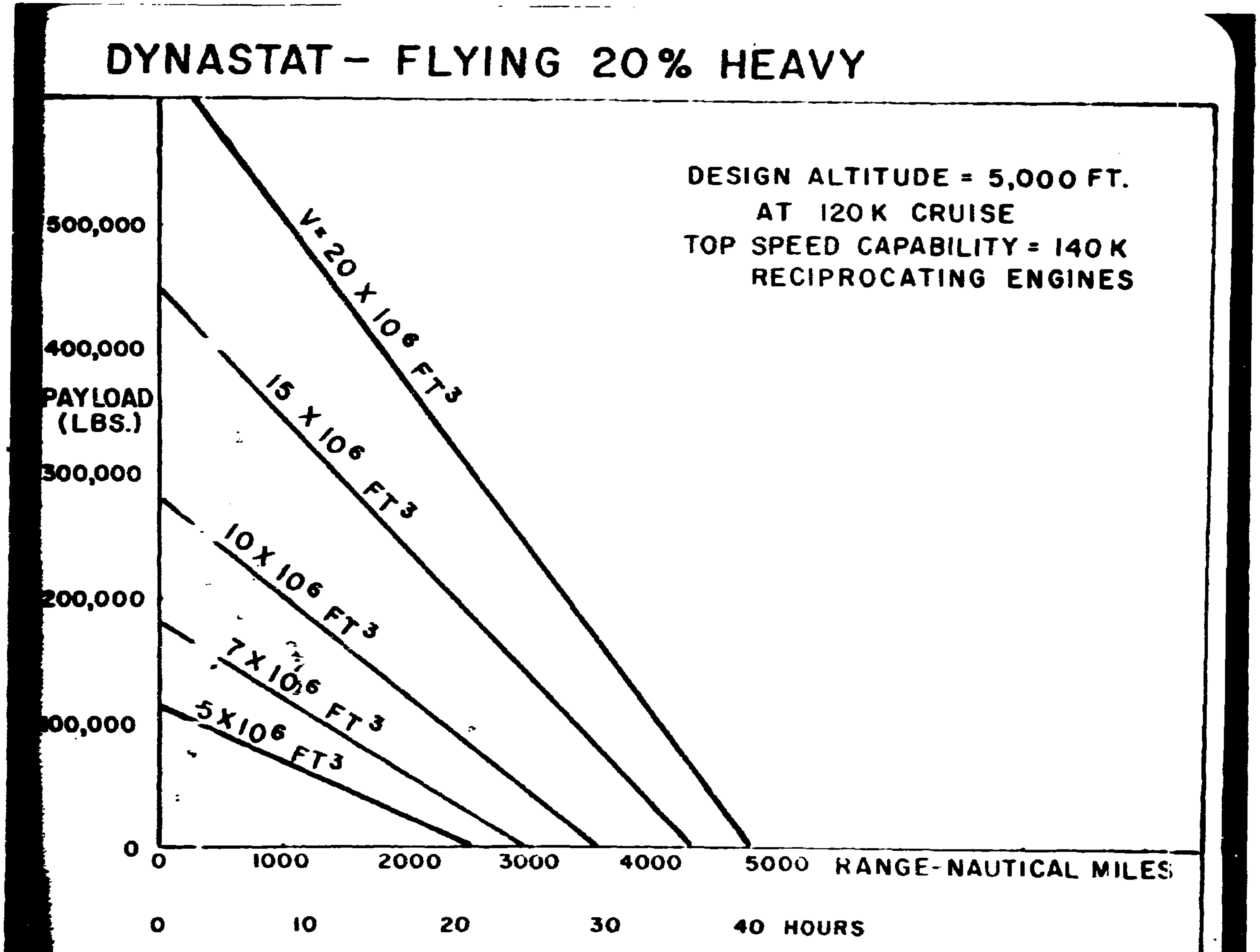

DYNASTAT – FLYING 20% HEAVY
DESIGN ALTITUDE = 5,000 FT.
AT 120 K CRUISE
TOP SPEED CAPABILITY = 140 K
RECIPROCATING ENGINES
V = 20 X 106 FT3
15 X 106 FT3
10 X 106 FT3
7 X 106 FT3
5 X 106 FT3
PAYLOAD (LBS.)
500,000
400,000
300,000
200,000
100,000
0
0
1000
2000
3000
4000
5000
RANGE-NAUTICAL MILES
0
10
20
30
40
HOURS

FIGURE 7

STATUS OF TECHNOLOGY

SMALL SILENT AIRSHIP	JOE II WORK GTR OPERATING EXPERIENCE LEAR WORK ON STEAM ENGINES
TRANSPORTER	STUDIES FOR NASA STUDIES FOR AIR FORCE REPLIES TO COMPANY REQUESTS
NUCLEAR TRANSPORTER	ORIGINAL BLIMP STUDY AIR FORCE PROPOSAL WORK WITH WESTINGHOUSE

FIGURE 8

AIRSHIP SIZE COMPARISON

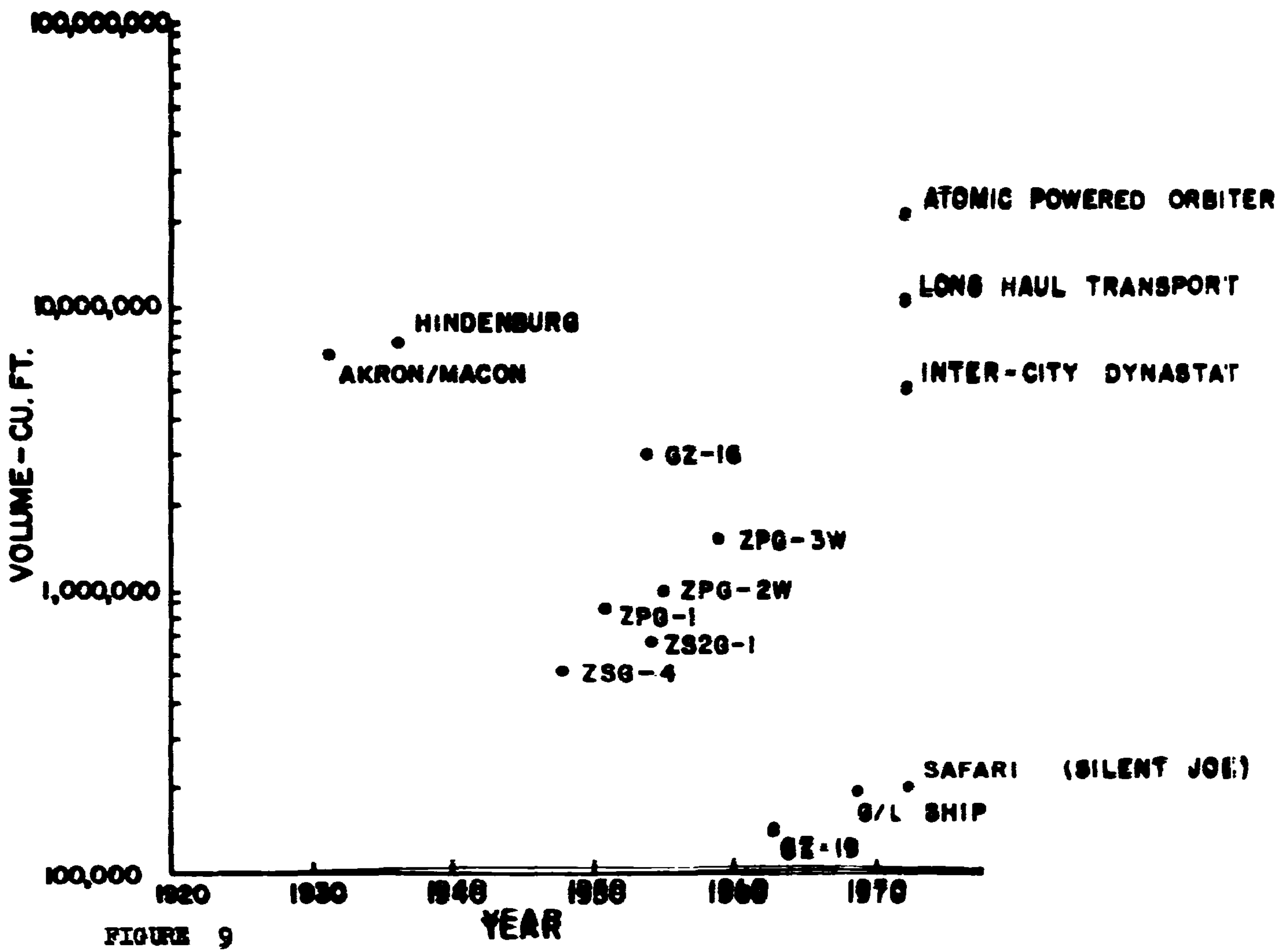


FIGURE 9

INTERMEDIATE CARGO TRANSPORT

VOLUME	5,000,000 CUBIC FEET
LENGTH	425 FEET
WIDTH	225 FEET
HEIGHT	100 FEET
DESIGN ALTITUDE	10,000 FEET
POWER PLANTS	20,000 HP - TURBO PROPS
TOP SPEED	140 KNOTS
USEFUL LIFT	160,000 LBS (WITH 68,000 LBS DYNAMIC LIFT USING TURBO-PROP POWER)

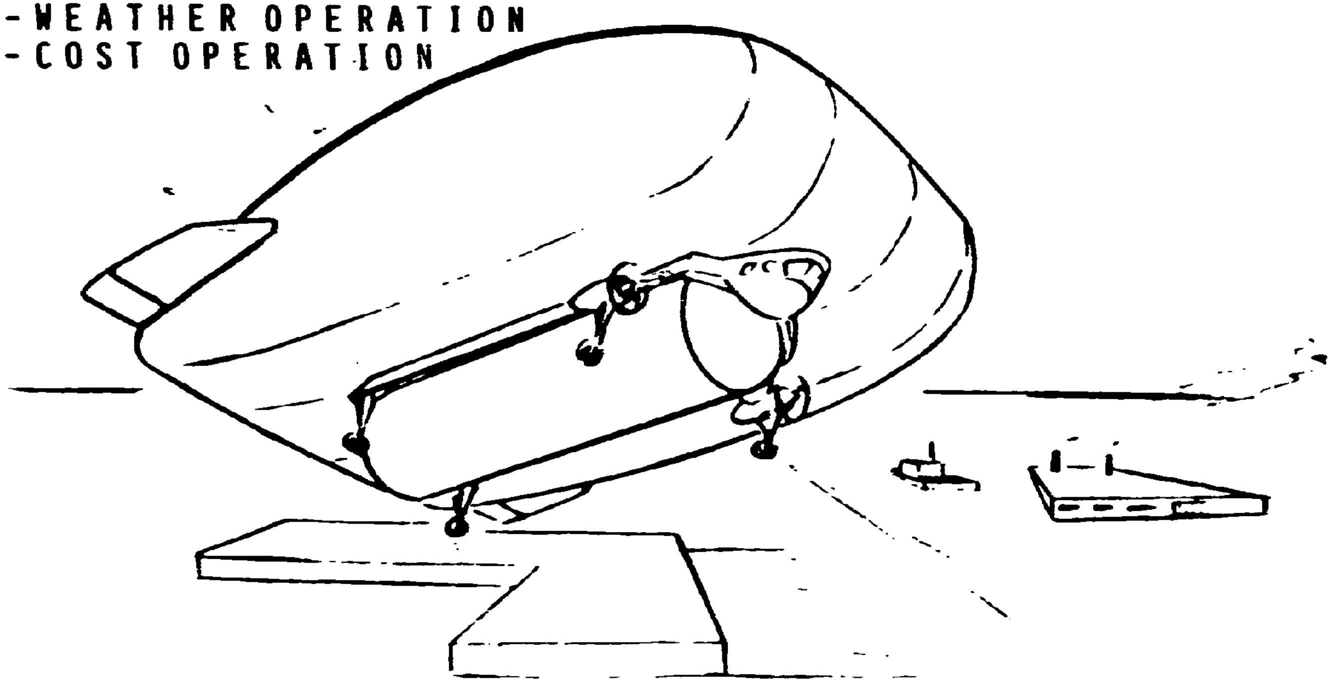
FIGURE 10

INTERMEDIATE CARGO TRANSPORT

(100 MILES PER HOUR)

POINT-TO-POINT DELIVERY
REDUCE CARGO VIBRATION
REDUCE CARGO SHOCK
ALL-WEATHER OPERATION
LOW-COST OPERATION

TRANSPORT COMPLETELY ASSEMBLED UNITS.



REACH INACCESSIBLE AREAS
SCHEDULE INTEGRITY
EXTENDED DISTANCE DELIVERY
V. T. O. L. - NO RUNWAY REQUIREMENT

MISSILE TRANSPORT
MINE OPERATIONS
FARM - LIVESTOCK

FIGURE 11

LONG HAUL-VTOL TRANSPORT

VOLUME - 10,700,000 CUBIC FEET

LENGTH - 730 FEET

DIAMETER - 175 FEET

DESIGN ALTITUDE - 10,000 FEET

POWER-PLANTS - 23,000 HP TOTAL

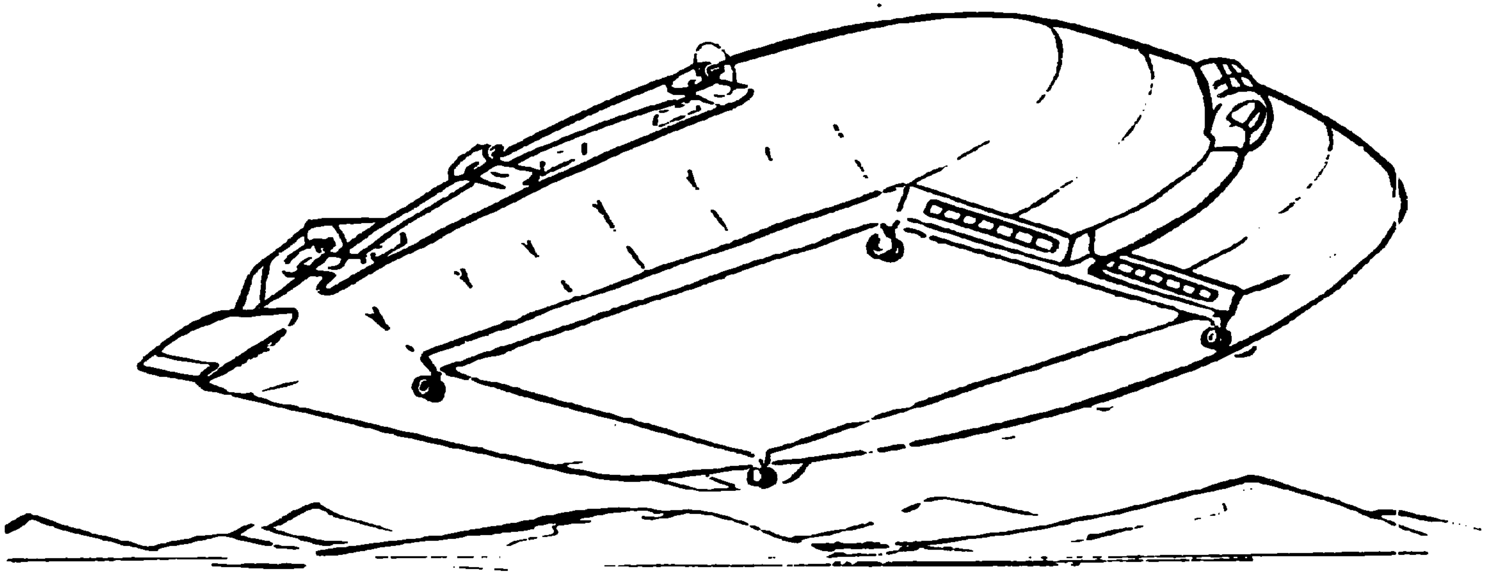
TOP SPEED - 140 KNOTS

USEFUL LIFT (WITH 125,000LB DYNAMIC LIFT = 275,000LBS.

FIGURE 12

GIANT TRANSPORT

(800,000-1,000,000 POUND PAYLOAD)
FUEL: CHEMICAL OR NUCLEAR



20,000,000 CUBIC FEET
200 M.P.H. CRUISE

WORLD POLICE - ARMIES - PASSIVE OBSERVATION - INTELLIGENCE - TELEVISION RELAY - AIRCRAFT BEACON

FIGURE 13

GIANT TRANSPORT

HOUSE-BRIDGES-FACTORIES-OBSERVATORIES

THE AIRSHIP IS FREE FROM HANDICAPS IMPOSED ON TRUCKS BY ROAD WIDTHS...... ON TRAINS BY TUNNELS AND TRESTLES. ON AIRPLANES BY SMALL LANDING FIELDS AND COMPARTMENT SIZES.

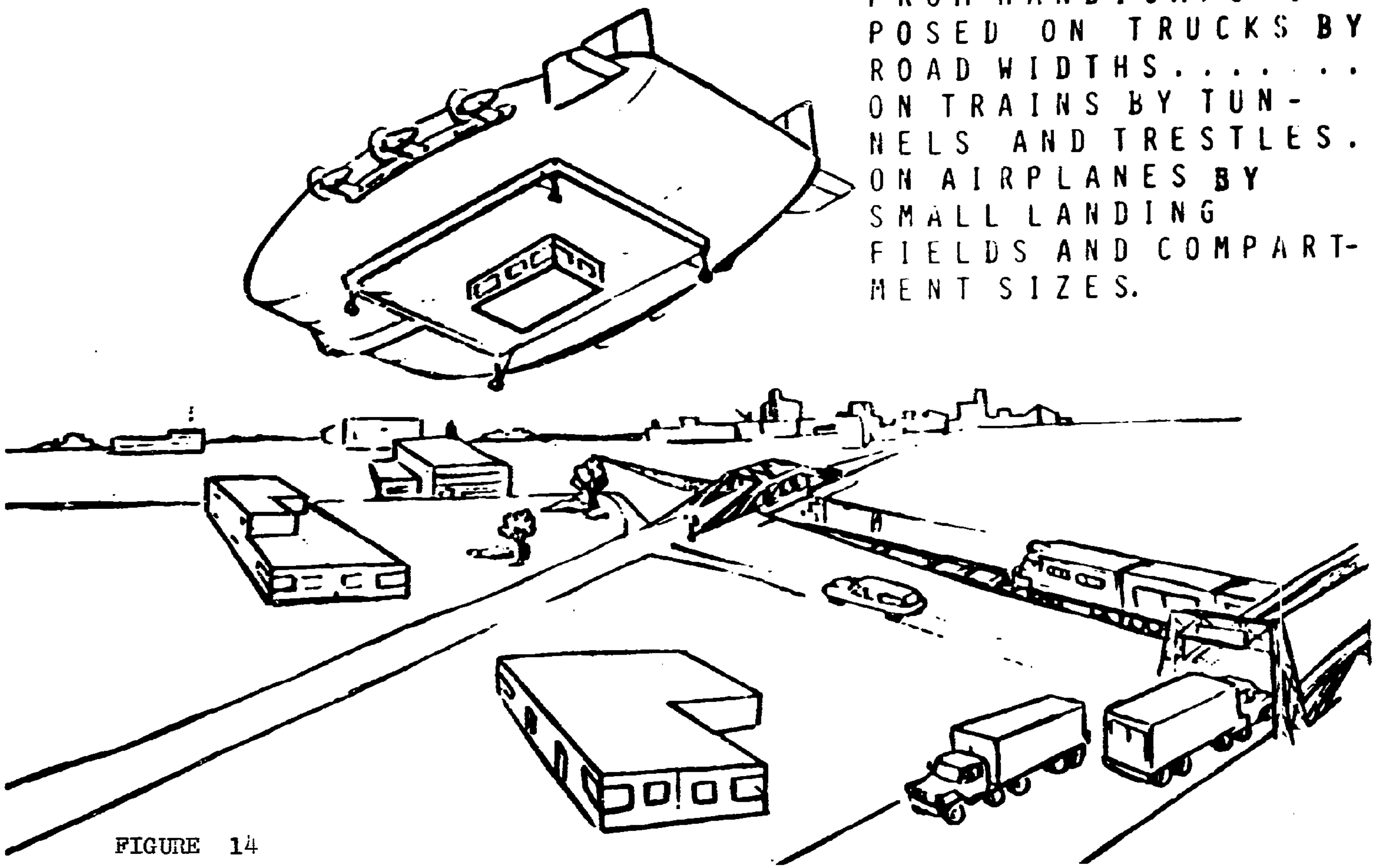
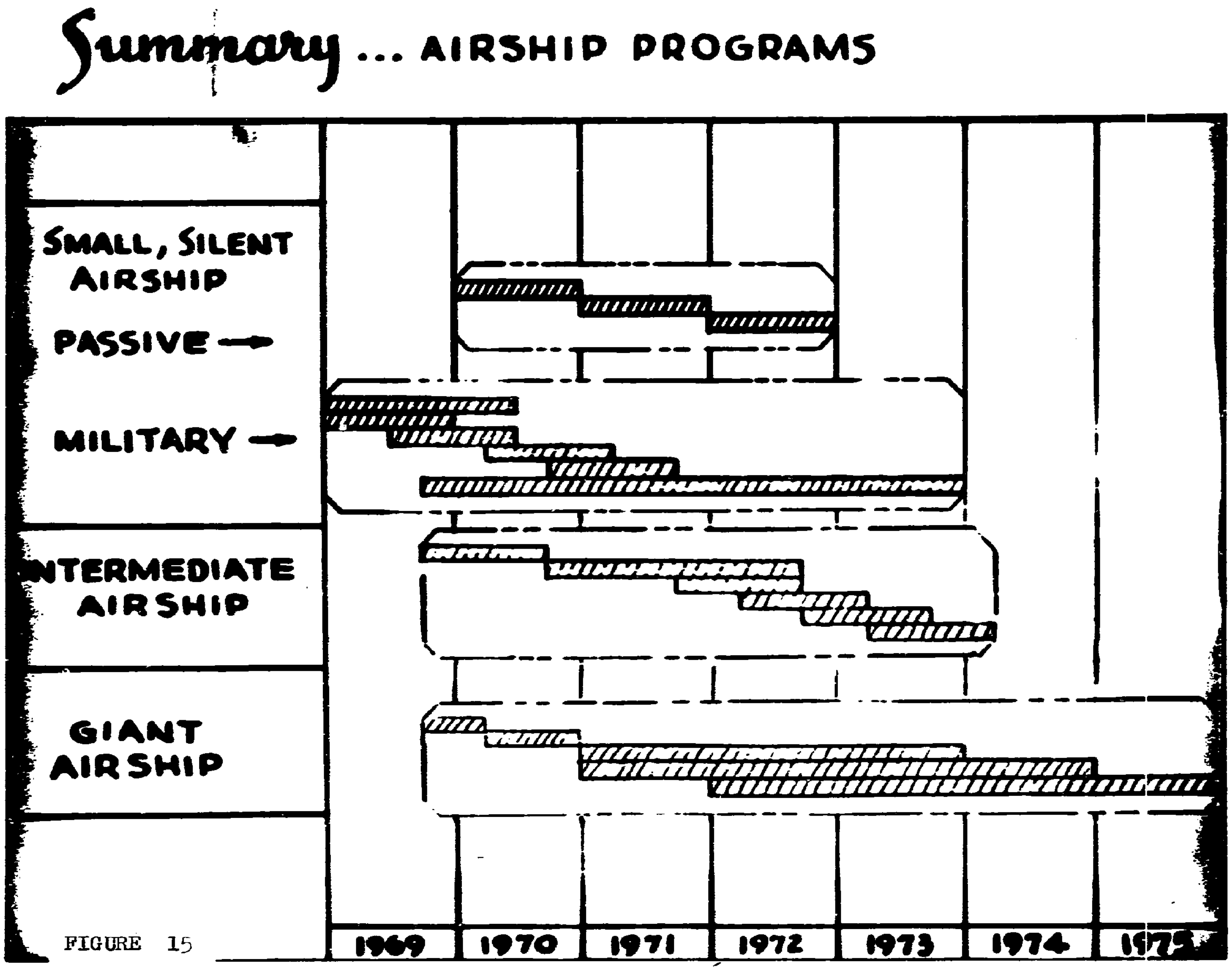
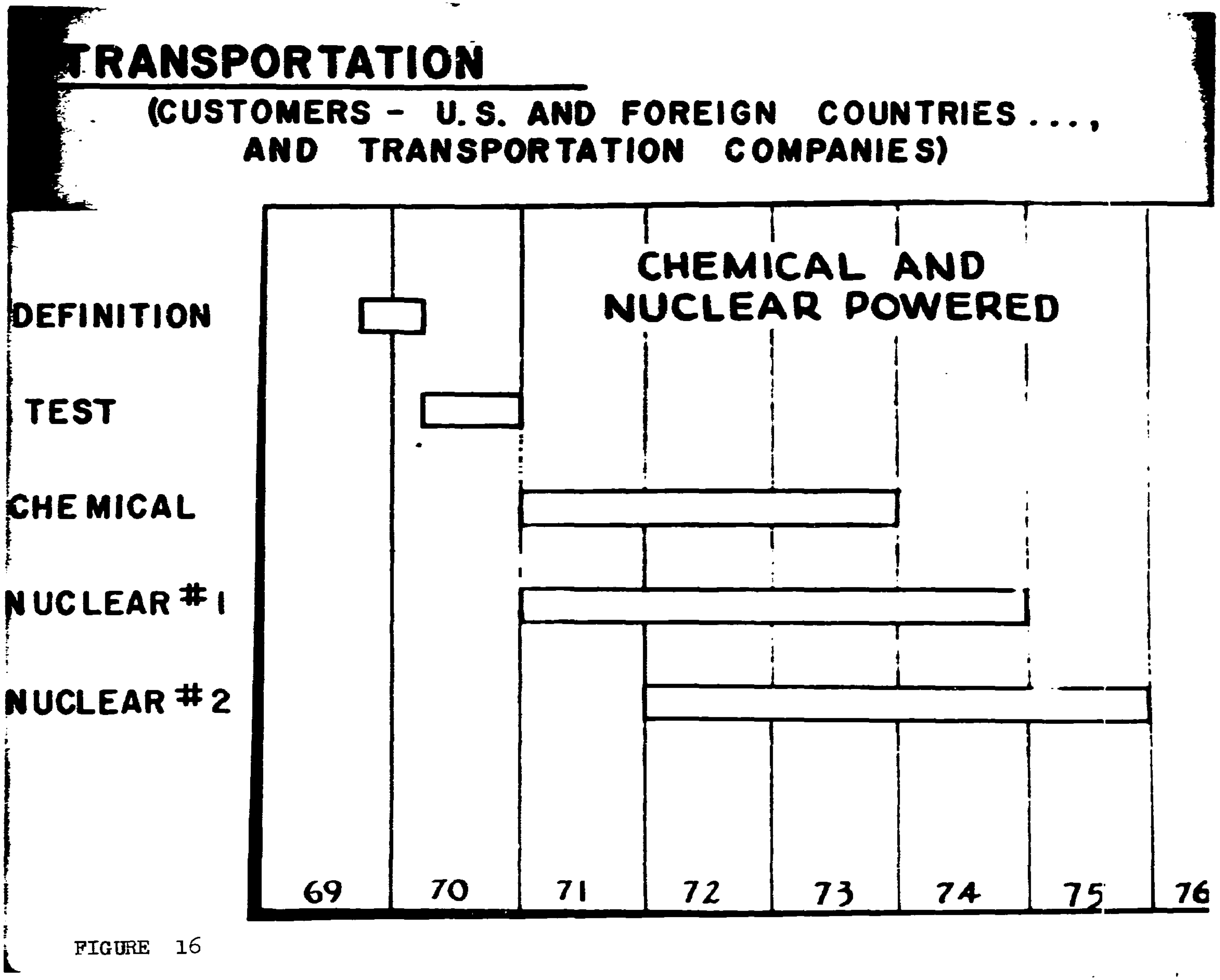


FIGURE 14



Summary ... AIRSHIP PROGRAMS
SMALL, SILENT AIRSHIP
PASSIVE
MILITARY
NTERMEDIATE AIRSHIP
GIANT AIRSHIP
1969
1970
1971
1972
1973
1974
1975

FIGURE 15



TRANSPORTATION
(CUSTOMERS - U.S. AND FOREIGN COUNTRIES....,
AND TRANSPORTATION COMPANIES)
CHEMICAL AND
NUCLEAR POWERED
DEFINITION
TEST
CHEMICAL
NUCLEAR #1
NUCLEAR #2
69
70
71
72
73
74
75
76

FIGURE 16

APPENDIX L

DESIGN ASPECTS OF EXTREMELY LARGE HELICOPTERS FOR THE VHLH

Presentation by John Schneider - Boeing-Vertol

In order to evaluate the potential of large helicopter concepts Vertol conducted a broad study to examine the influence of propulsion systems on configuration layouts, payload capabilities, and performance of extremely large helicopters. The investigation was based on the following ground rules: 1970 technology, payload range of 20 to 70 tons, hover capability out of ground effect (4000'/95°F at military rated power), maximum disc loading of 10 pounds per square foot, maximum speed less than 150 knots, mission radius of 50 nautical miles and hover time of a half an hour.

Before looking at the feasibility of extremely large helicopter designs, it would be desirable to take a look at helicopter progress as a firm basis necessary for the extension of our design technology. Figure 1 illustrates the growth of large helicopters over the last 25 years. Except for the FA-284, which was powered by reciprocating engines, the upper bound of the growth trend was set by turbine-engined aircraft. Shaft turbines have sparked these developments with one exception, and that is the XH-17 which utilized compressor bleed from twin turbojets plus rotor tip burning. Although US helicopters have progressed rapidly, they have been completely overshadowed by the Russian Mil developments, the Mil 6 and Mil 10 single rotor helicopters and the latest, the Mil 12 lateral twin rotor type. Figure 2 compares the payload capability, for a hover out of ground effect at 4000'/95°F, and the physical size of some of the pacesetting helicopters. It can be seen that as hovering vehicles, the Mil 10 and Mil 12 do not compare with the CH-47 and the CH-54. The Mil 6 and Mil 10 are about twice the gross weight, and yet they do not have much additional payload. However, the Mil aircraft do indicate the feasibility of extremely large helicopters in the 50-ton payload class since the physical size of their propulsion components are representative of those required by the more efficient helicopters that are being considered here.

Probably the two most important factors driving the various approaches to a very large helicopter are those of rotor torque and rotor diameter. As gross weight increases, rotor diameter increases at constant disc loading. In the absence of certain constraints, it would be desirable to increase tip speed with diameter; however, profile power losses in hover and Mach number effects in cruise, as well as noise problems, limit tip speed to something below 800 feet per second, probably 750 feet per second. Therefore, as the rotor diameter increases much beyond that of today's aircraft, the rpm of the rotor decreases linearly with diameter resulting in higher transmission design torque, higher unbalanced torque, and most importantly a lower centrifugal force to blade lift ratio which leads to coning and droop problems.

Now these fundamental factors have promulgated most of the many past and present rotor drive system concepts; however, these effects have been relieved in the previous helicopter generations by a gradual increase in tip speed from about 500 feet per second in the late '40s to 700-750 feet per second today. Having now reached the probable limit in tip speed, for noise reasons if nothing else, we have to re-examine the effects of such a limit on future single and multi-rotor concepts. Figure 3 illustrates 5 representative large helicopter concepts sized for a gross weight of 350,000 pounds (at a disc loading 10 psf). All of these concepts have certain redeeming features ranging from a minimum number of components, as in a single rotor, to that of a minimum size propulsion elements, as in the quad rotor. In the following will be considered all of these feasible helicopter concepts including what is not shown here, the co-axial.

Helicopter propulsion systems are variations of either shaft-drive or tip-drive rotor concepts. To date, successful production helicopter programs have been based either on reciprocating engines or turbine engine shaft-drive systems. However, the past 30 years have seen an almost continuous exploration of tip-drive variants aimed at, among other things, the elimination of the rotor gearbox. These tip-drive concepts have varied from propellers on rotor, ram jet, pulse jet, tip turbojet, cold cycle with or without tip burning, and, so on, to the warm and hot cycle jets that are under investigation today. Great promises have been held out for the tip-drive systems. The problem has been that the prediction of the airframe weight savings have been more than negated in the past by large fuel flow requirements. However, in order to fully understand the fundamentals of the available options for future large helicopter propulsion systems, this study includes both warm and hot cycle drive systems for a single rotor as well as shaft-drive systems as shown in Figure 4.

In order to look at the influence of propulsion system elements on extremely large helicopters, we have to look at all the elements independently and then combine them to get an overall position on the relative system merits. So, first, looking at the powerplant - It is well recognized that aircraft powerplant technology has improved remarkably since the advent of the helicopter. The change from reciprocating engines to gas turbines was as great a breakthrough for helicopters as it was for transport airplanes. Although the early gas turbine did not match the reciprocating engine in fuel consumption, its good specific weight, plus compactness, smooth torque output, low oil consumption, lower noise, and especially the free turbine output shaft, have more than compensated. The past several years have seen the introduction of the so-called advanced technology engines, primarily in the area of turbofans. They are called advanced technology because of higher pressure ratios, like 20 to 1, high turbine temperature like 2200 to 2400°F, and advanced materials and structural designs. These core engines have introduced a completely new range of turbojets and turbofans for the airplane market. Derivative shaft-turbine versions have been proposed but none of them have been procured due to the lack of a helicopter or V/STOL program. Figure 5 compares the characteristics of today's shaft-turbine engines with the advanced technology derivatives, as well as proposed new engines. New engine programs offer certain advantages, like slightly better SFC's and somewhat better power to weight ratio, but this must be balanced against their increased cost. Derivative engines may have slightly higher SFC's, and weight, but this must be balanced against their lower development and procurement costs. In addition, the value of higher TBO's and MTBR's resulting from common component experience cannot be overlooked. Advanced technology derivatives may indeed be more cost effective than new engines, at this time, providing that a power match is possible. Figure 6 matches shaft-turbine engine power required to power available for certain derivative engines. It can be seen that, generally speaking, on a shaft-drive helicopter that a match can be obtained with no more than four engines.

In Figure 7 are advanced technology derivatives available in fan engines; a match can be obtained here with no more than 2, 3, or 4 gas turbines.

In addition to weight characteristics, a primary parameter for a successful aircraft is the drive efficiency, which is the ratio of mechanical rotor power delivered to the gas power available. In Table I can be found the incremental losses for each of the systems, at a typical tip speed of 750 feet per second. In the shaft-drive system, a power turbine is added to the core engine to provide shaft-power output. Power turbine losses are about 12 percent with an additional 6 percent loss due to residual exhaust kinetic energy. Power is then transmitted to the rotor shaft through a gearbox and shafting arrangement, with losses depending on number of meshes and gear characteristics. For equivalent designs, these losses are about 2.7 percent for either single or multi-rotor aircraft. The single rotor shaft-drive system requires additional power for main rotor torque compensation amounting to an additional 8.5 percent loss.

In the hot cycle and warm cycle tip drive systems, the gas from the core engine (and fan) passes through a ducting system to the rotor tip cascade nozzle. Careful attention must be paid to the ducting elements in order to achieve an efficient system. The mixer/diffuser, diverter valve, sliding and rotating seals, inside surface roughness, etc., can contribute significant losses. For this analysis, minimum losses are assumed based on the expectation that pressure and temperature gains from centrifugal pumping will compensate for these losses.

The left side of Figure 8 summarizes the variation in Drive System Efficiency with main rotor tip speed for the systems considered. The warm cycle system shown here was based on an optimization of bypass ratio, external aerodynamics and blade materials requirements. Since this study was based on equal gas horsepower available and therefore a constant fuel flow, the specific fuel consumption is inversely proportional to the drive efficiency. Therefore, the warm and hot cycles will use fuel at a ratio of 1.5 and 2.1 times that of the best shaft-drive as shown on the right side of Figure 8. One of the trade offs available to offset the poor fuel consumption is the elimination of much of the drive system weight needed for the shaft-drive system.

Rotor transmissions are the primary element in the shaft-drive system. The other gearboxes in the system usually operate at relatively high input and high output speeds, and because of the low torque can, therefore, transmit large powers for relatively low weights; however, the rotor transmission must reduce a relatively high speed power input to a low speed of the rotor. Because of the rotor tip speed limitation discussed earlier, rotor rpm for the large helicopters discussed here will decrease linearly with diameter. Figure 9 indicates the trend of rotor transmission design torque versus design gross weight for both single and multi-rotor systems. Substantial progress has been made in the development and manufacture of large gearboxes. Facilities now exist for machining, grinding, and heat treating gears to such size as to accommodate nearly a million and a half foot pounds of design torque. For this amount it can be seen that in a twin rotor system you can get out to about 300,000 pounds gross weight and in a tri rotor/quad rotor you can go on to 400,000 pounds. Development of the tip-jet systems began in the days when the shaft-drive systems weighed about one pound per horsepower. However, drive system weight has improved to today's value of about six tenths pound per horsepower. It is anticipated that a reduction of about 20 percent is possible due to projected technological advances in materials, manufacturing processes, lubrication media, and new tooth forms. Figure 10 contrasts the shaft-drive systems weight with that of the hot and warm drive elements. For the shaft-drive system these elements include all the gearboxes, shafting, couplings, lubrications systems, and so on. For the hot and warm cycle systems the elements include the mixer/diffuser ducting, diverter valves, hub transfer ducting, rotor blade sliding and rotor joints, main rotor shaft, power takeoff for the yaw rotor, and shafting and gearboxes in the yaw rotor system.

Let us now look at another propulsion element, the rotor. First, we will look at conventional rotors. As indicated earlier, some of the most influential factors in the development of extremely large helicopters are rotor systems technology and performance limitations. Due to fundamental limitations on tip speed, increasingly large rotor diameters will suffer weight penalties in order to integrate the detailed design requirements imposed by flapping loads, centrifugal force, torsional rigidity, ground resonance, rotational inertia, droop criteria, and the coning criteria. The primary cause can be attributed to a decreased ratio of centrifugal force over blade-lift or as called, loss of centrifugal relief. Figure 11 illustrates this effect on the blade coning angle in hover. Although a moderate blade coning angle in itself is no problem during hover, its effect is felt in a cruise mode when flapping due to the velocity differential, maneuvers, and gusts is impressed on a high steady coning angle. Experience has shown that to avoid high vibration and increased power requirements in cruise flight, steady coning angles should be no larger than those shown in this somewhat ill-defined gray area of Figure 11. Of course, tip weights or outboard structural weight may be added to reduce coning, but this is at the expense of static droop angle and increased empty weight. Increased droop angles require larger pylons, longer landing gears, or special droop stop mechanisms. Therefore, increased blade thickness inboard is generally required for droop allevation. The net effect of the droop and coning control requirements is to cause the rotor system weight to increase at a faster rate for rotor diameters above 80 to 90 feet as shown in Figure 12. The weight of present US production rotor systems is approximately proportional to diameter to the 2.18 power. However, the weights of rotors above 80 to 90 feet in diameter will be approximately proportional to diameter to the 3.2 power.

It is possible that future advancements in new materials and fabrication techniques will provide some relief to the droop problems, thereby allowing increased weight to be available for the coning solutions. The advanced fiberglass blades as well as boron/fiberglass blades are the subject of high priority programs now underway.

Now looking at rotors for tip jet drive helicopters, they have fundamental constraints of their own as well as those of conventional rotors. Some of their constraints are the materials and design concepts to provide for warm or hot gas ducting; reduced figure of merit of 13 to 20 percent due to increase in thickness ratios and very low blade aspect ratios; the problem of matching external aerodynamics and gas flow volume requirements; the vulnerability or safety aspects due to rupture or puncture of the hot gas ducting; control loads due to necessity of unbalanced blades to avoid large chordwise balance weight penalties; and control stiffness requirements due to unbalanced blades. Although the high torsional stiffness of the large warm and hot cycle blades preclude flutter of the structure, the unbalance requires very stiff control system linkages for flutter avoidance. The effects of most of these factors is reflected in an increased rotor weight fraction for both warm and hot cycle systems. Figure 13 correlates previous tip jet rotor system weights on a parametric trend basis with those of conventional articulated rotors. Although lower weights for hot cycle rotors have in the past been put forward, this figure is believed to show a representative family trend. Now this conclusion is further strengthened by the data on this next curve, Figure 14, which shows the relationship of certain tested tip-jet rotors and rotor design studies to existing conventional rotor weights.

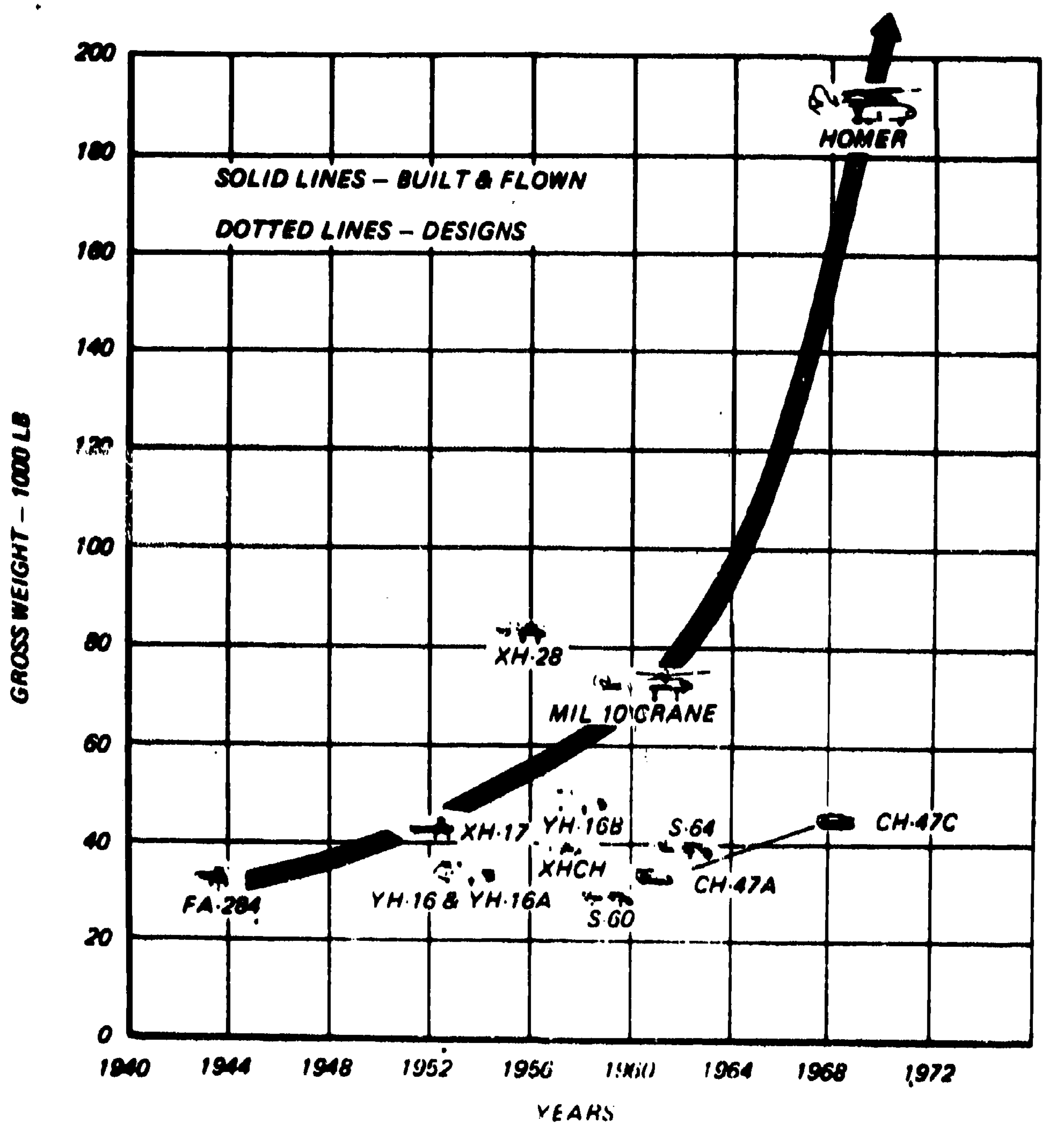
The tip-jet rotor does not (in the foreseeable future) have a coning problem. The increased rotor weight, including the ducting and tip cascades, provides adequate coning control for diameters up to about 250 feet. The increased thickness ratio required for gas flow volume may reduce droop angle although this effect is somewhat less due to the outboard weight distribution.

The influence of the propulsion system on extremely large helicopter design can now be summarized by examining its effect on both weight and performance. Other factors, such as technical risks, costs, and timeliness, are strongly related in this case to the aircraft sizing resulting from payload requirements. The configurations in this study were identical in philosophy, except as modified to reflect the requirements of the specific propulsion concept. Figure 15 presents the component weight breakdown for 8 of the concepts investigated for a design gross weight region of 100,000 to 350,000 pounds. Generally, the large differences in concepts are a result of rotor system, drive system, and mission fuel weights that follow from the previous discussions. The significant points to be noted are the values and slopes of the payload fractions. Although payload fractions of both warm and hot cycle are small, the slope indicates little variation with gross weight. However, the shaft-driven single lifting rotor because of the rotor and drive system weight effects discussed earlier, loses its payload carrying capability with increased size. The co-axial follows the same trend although delayed due to lesser drive system effects. The shaft-driven multi-rotor systems show the best payload fractions with less effect from increased gross weight. This effect has been recognized by the noted designer of very large helicopters, Dr. Mil, as he noted in his recently published book, and has resulted in his switch from the large single lifting rotor to twin rotors in order to delay the weight escalation while providing a higher payload capability. Figure 16 converts the same payload fraction data into payload available trends as a function of design gross weight. In summary, this figure shows the potential of reaching the 50 ton or greater payload goal with the next generation multi-rotor helicopter. In fact, at 350,000 pounds gross weight in a multi-rotor, a 60-ton payload can be obtained. The shaft-driven single lifting rotor does not match this capability and the warm and hot cycle concepts appear to be competitive only at gross weights above 400,000 pounds.

Now looking a little into cost effectiveness, the criterion for measuring the economy of various transport aircraft is generally based either on the cost of procurement or on operational cost criteria. Cost of procurement correlates with the empty weight of the aircraft. A common index of procurement economy in engineering terms is, therefore, the ratio of productivity to empty weight, as shown in the left side of Figure 17. In the field, under wartime conditions, expenses incurred in the past are no longer of consequence so the cost of fulfilling the immediate task is determined mainly by the amount of fuel consumed. Thus, the economy of field operations can be expressed as a ratio of productivity to fuel required as shown in the right side of Figure 17. A cursory glance at these two curves will show that the multi-rotor systems are relatively more attractive for future large transport helicopters.

Primary technical risks in the development of these large payload helicopters appear to be in the area of rotor and drive system technology especially for the single, co-axial, warm, and hot cycle concepts. The use of multi-rotor shaft-drive systems alleviate these risks to a large degree in a lower payload range (like 20 to 40 tons). For payloads of 40 tons or more, a multi-rotor concept is clearly required. Figure 18 summarizes the significant sizing risks in terms of rotor diameter, rotor gearbox design torque, and rotor gearbox weight. Note the diameters of actual rotors on this left-hand chart. For the shaft-drive single rotor, the significant risks are in rotor development for very large diameter rotors and manufacture of very high torque gearboxes. Aside from these, the concept is well understood having had many years of production and operational experience. For the warm and hot cycles we have in addition to rotor development risks for the very large rotor diameters required, the need for development and operational test of adequate structural material concepts for the blade; vulnerability aspects due to the rupture or puncture of the hot gas ducting require testing under actual operational conditions. In the co-axial rotor area, although the rotor gearbox design torque risks are reduced from those of single rotors, the development of the very large rotor and large gearbox are the significant technical risks. Although, there have been co-axial production programs, these have been confined to relatively small aircraft. In tandem and lateral twin rotor concepts the development risks do not appear to be of major significance in either the rotor or drive system area. The tandem rotor helicopter has a record of many years of production and operational experience. The lateral twin lacks this history of problem solving; therefore, there may be fundamental risks in the lateral twin that are not yet well understood, such as vibration and flying qualities. In the tandem, desirable improvement in vibrations, flying qualities and so on have been identified and solutions are available. In the tri-rotor and quad-rotor concepts, obviously, serious rotor and gearbox risks are practically nonexistent in either of these concepts. However, neither concept has an operational history. A previous AGARD study indicated problem areas in decreased mechanical reliability, increased structural complexities and structural weight.

In conclusion, it is evident from the foregoing that the goal of the 50-ton or greater payload capability is possible in the next generation helicopter providing that a multi-rotor concept is selected. Neither the shaft-driven single lifting rotor, co-axial, warm cycle or hot cycle concepts show an economical potential for this payload capability. Although, multi-rotor concepts, such as the tri-rotor and quad-rotor would have lower technical risks in the rotor and gearbox systems, increased structural complexities and structural weight, as well as decreased mechanical reliability detract from their potential. These effects must be traded off against the desire for order-of-magnitude improvements in maintainability. Twin rotor systems to be considered are the lateral twin and tandem concepts. Although, several of the early helicopters, and the recent Mil 12, were of the lateral twin layout, the concept has not yet had the benefit of an extensive service history to work out any unknown problems. Therefore, a lateral twin concept may be considered a greater risk than the tandem concept. In the tandem rotor concept, the technology base developed as a result of over 2 million flying hours in civil and military operations will provide the improvement necessary for a successful large helicopter development.



Growth of Large Helicopters

FIGURE 1

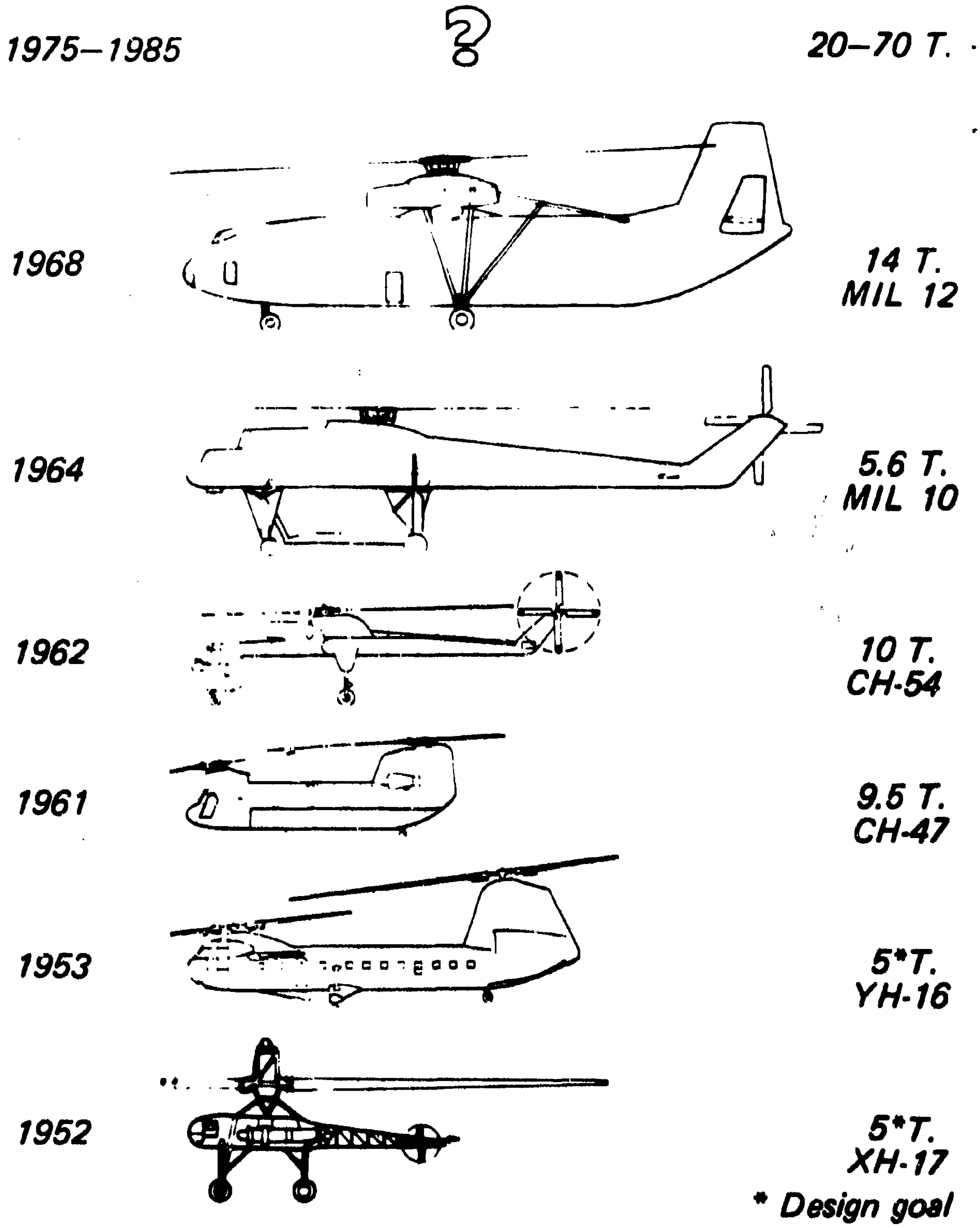


FIGURE 2 *Helicopter Payload Capability*

200-14

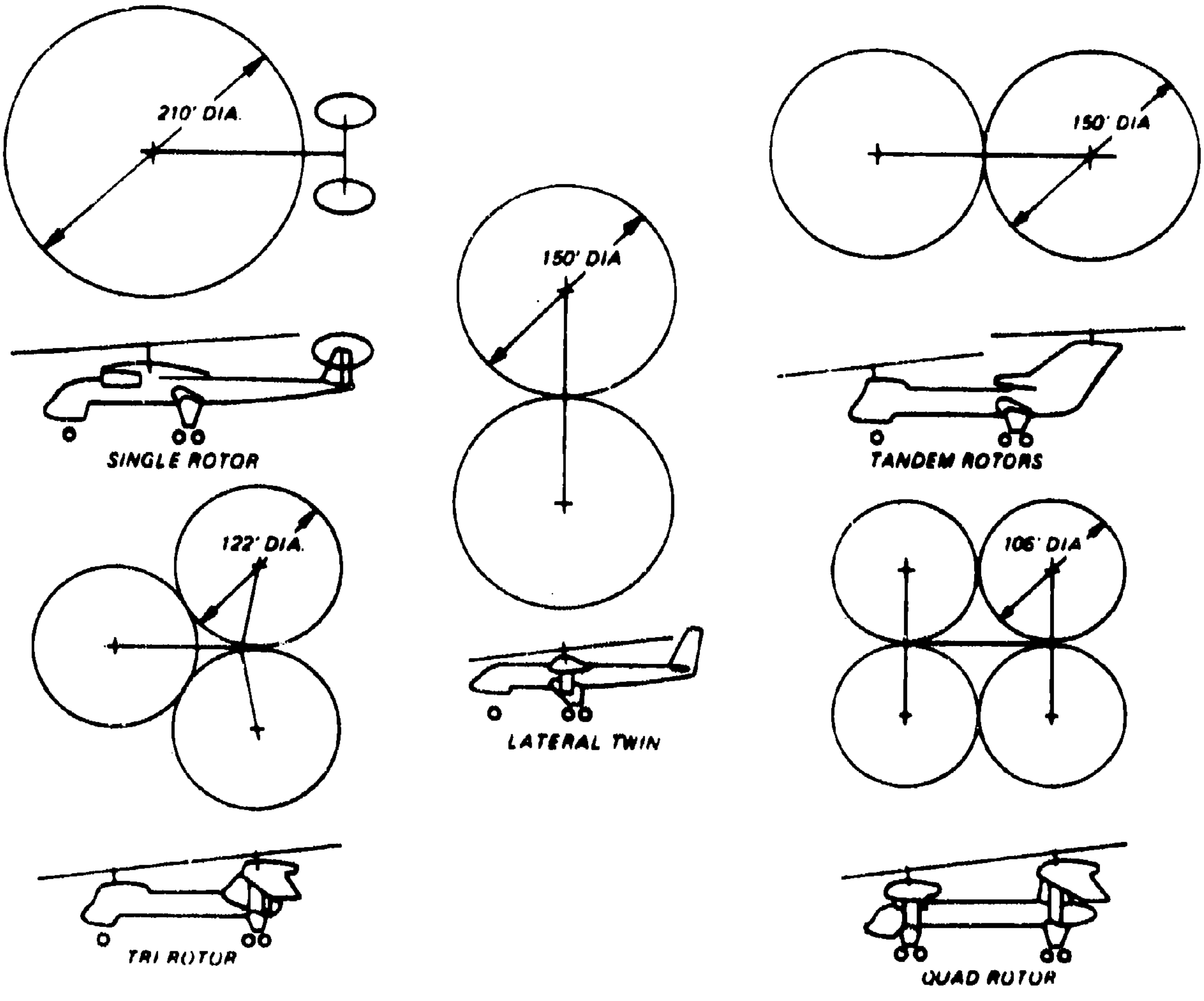
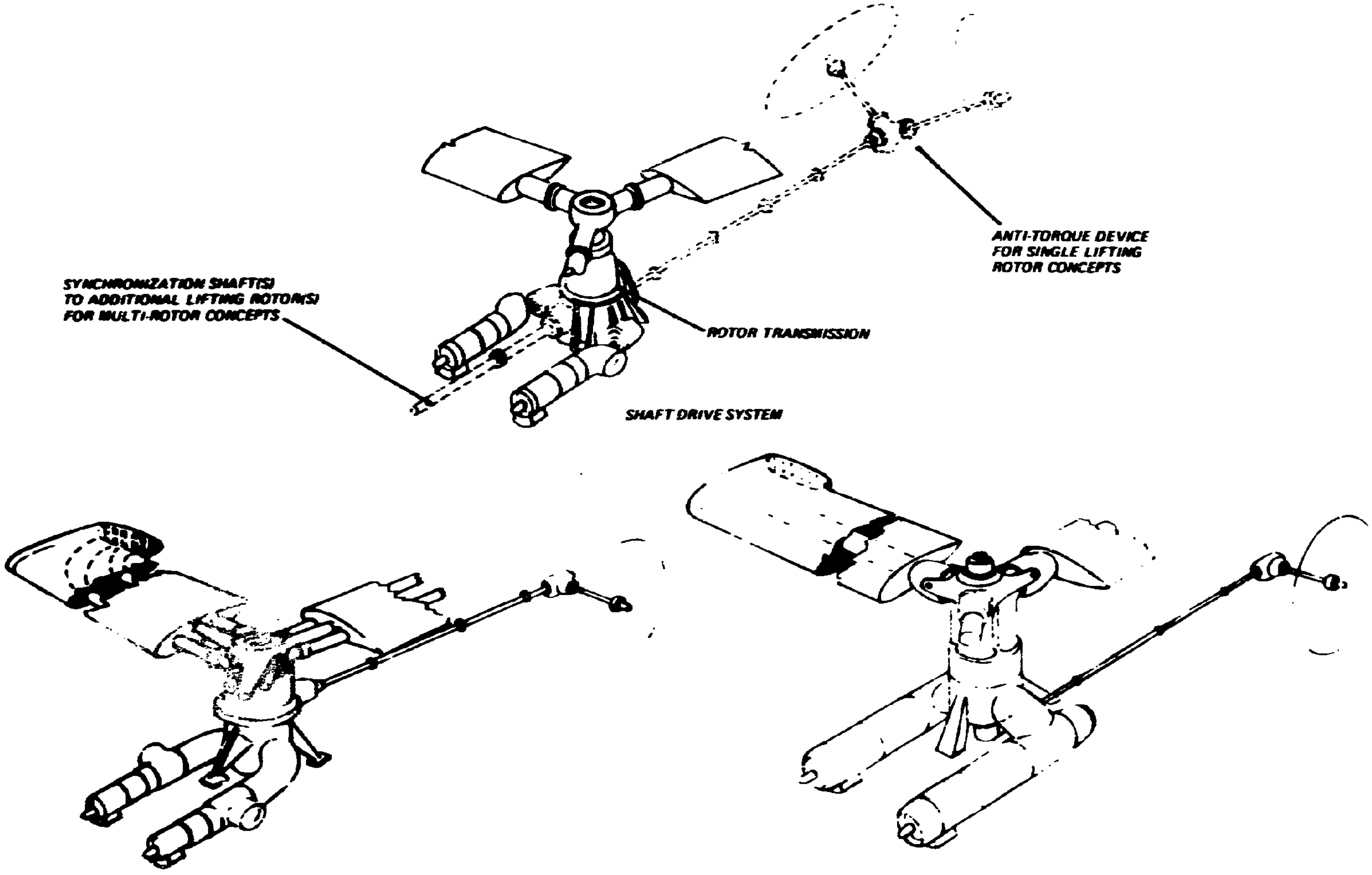


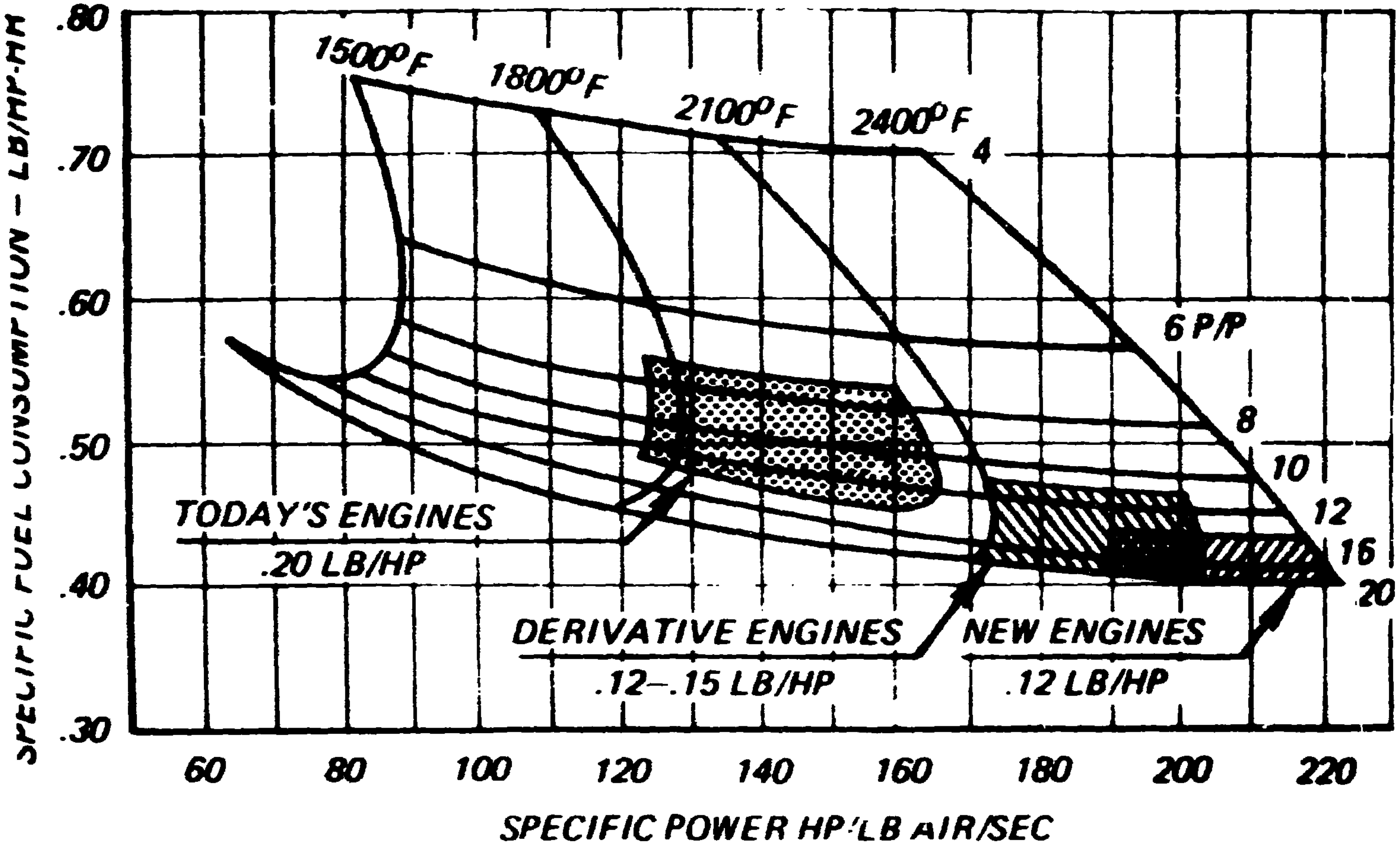
FIGURE 3 *Potential Large Rotor Combinations (Gross Weight 350,000 Disc Loading 10)*



Propulsion Systems

FIGURE 4

200-14



Shaft Turbine Characteristics

FIGURE 5

S/S

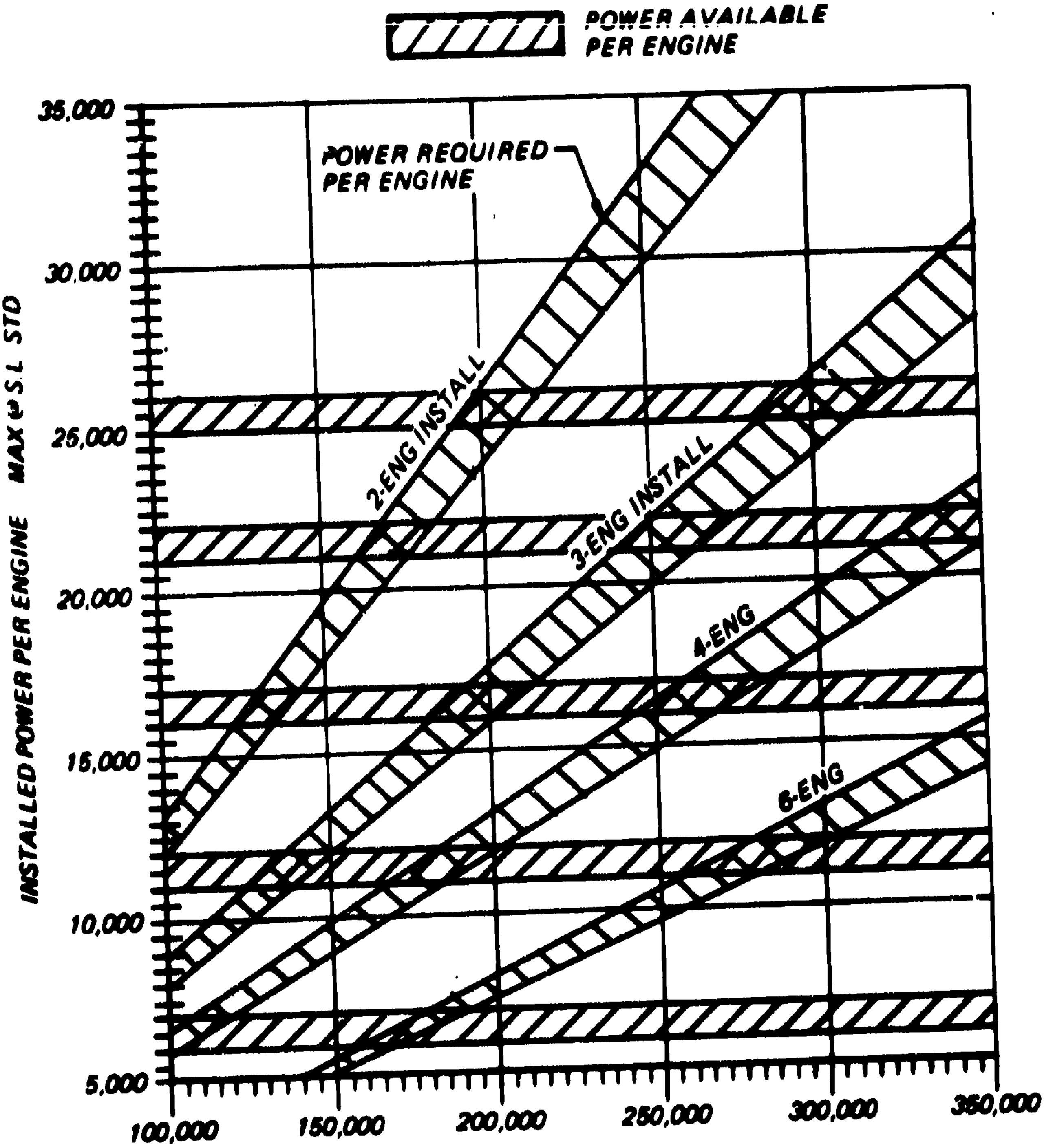


FIGURE 6 *Shaft Turbine Engine Requirements/Availability*

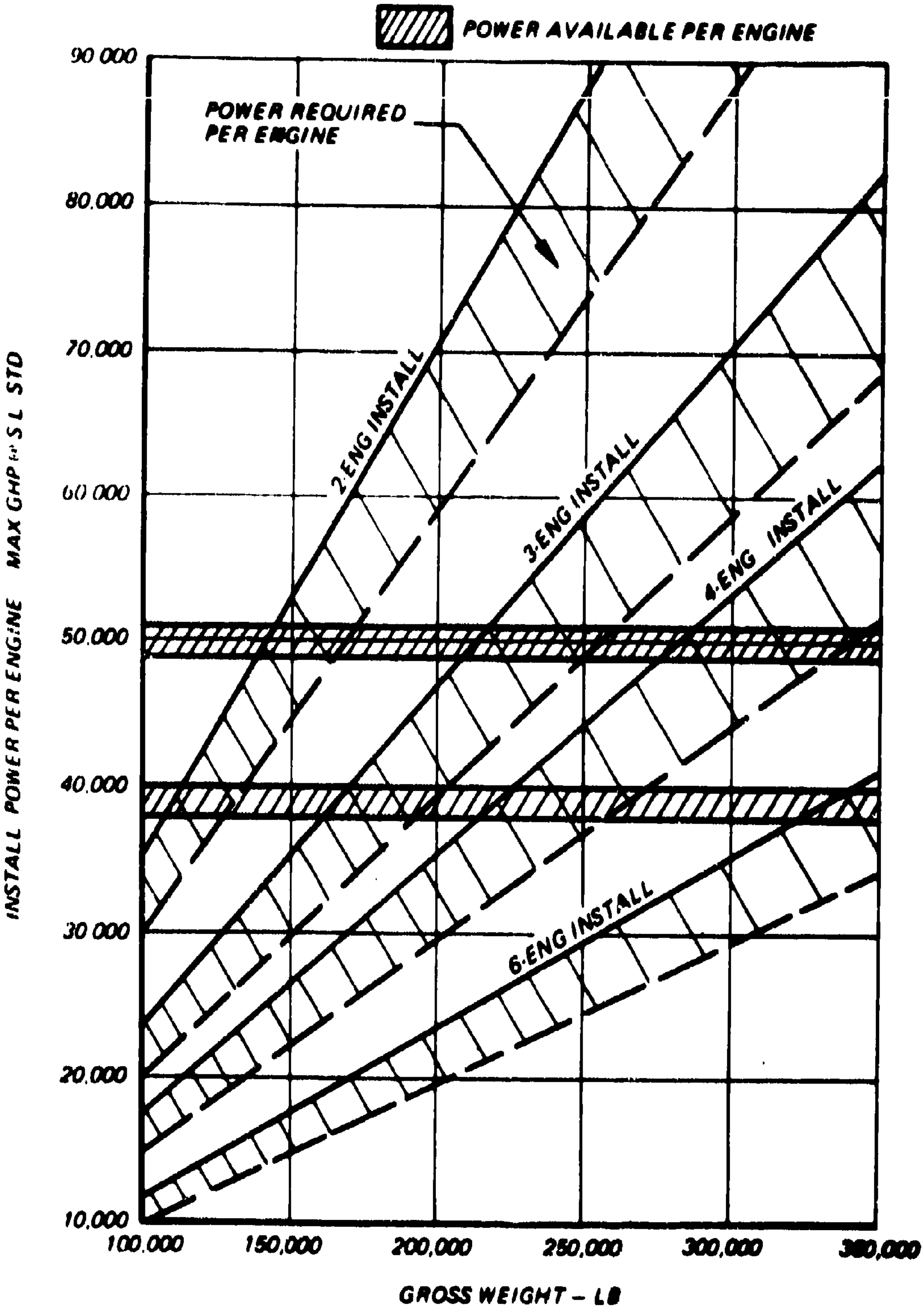
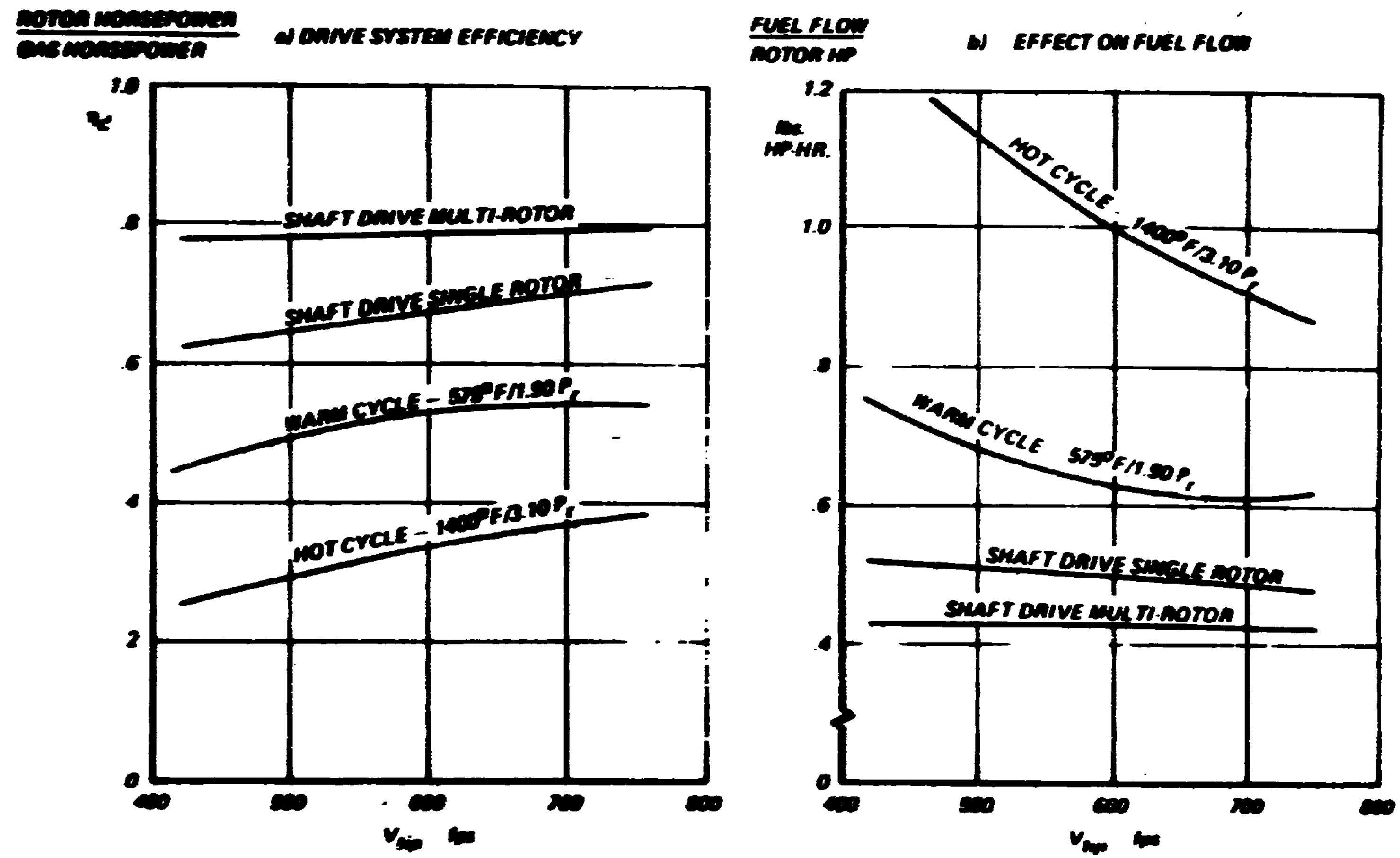


FIGURE 7 Hot and Warm Cycle Gas Generator Requirements/Availability

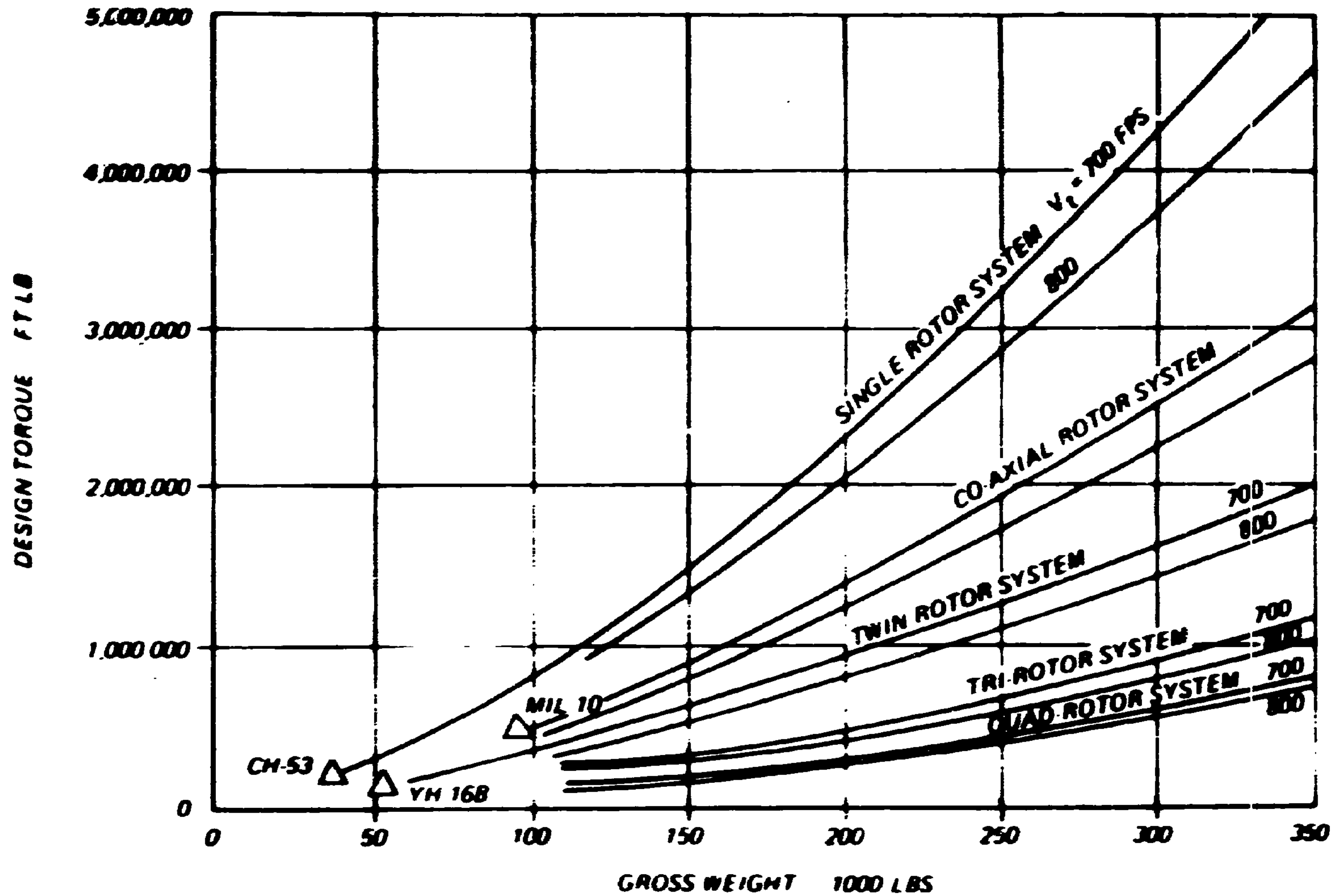
TABLE I
DRIVE SYSTEM EFFICIENCY
(4000 -95°F; DISC LOADING - 10 psf; TIP SPEED 750 fps)

	SINGLE ROTOR — SHAFT DRIVE	SINGLE ROTOR — HOT CYCLE	SINGLE ROTOR — WARM CYCLE	CO-AXIAL, TWO/FOUR ROTORS — SHAFT DRIVE	TRI-ROTOR — SHAFT DRIVE
I. SYSTEM PROPULSION LOSSES	.292	.62	.46	.207	.21
A) POWER TURBINE	.12	—	—	.12	.12
B) EXHAUST K.E.	.06	—	—	.06	.06
C) DUCTING-FRICTION*	—	.025	.025	—	—
-TEMP.*	—	.015	.015	—	—
D) CASCADE*	—	.02	.02	—	—
E) GEAR BOX	.027	Neg.	Neg.	.027	.027
F) TORQUE COMPENSATION	.085	—	—	—	.003
G) TIP JET EXPANSION AND KINETIC ENERGY LOSSES	—	.62	.46	—	—
II. SYSTEM DRIVE EFFICIENCY (1.00 - TOTAL ITEM I)	.708	.38	.54	.793	.79



Drive System Efficiency and Effect on Fuel Flow (4000 – 95° Disc Loading = 10)

FIGURE 8



Rotor Transmission Design Torque
(Disc Load = 10 psf)

FIGURE 9

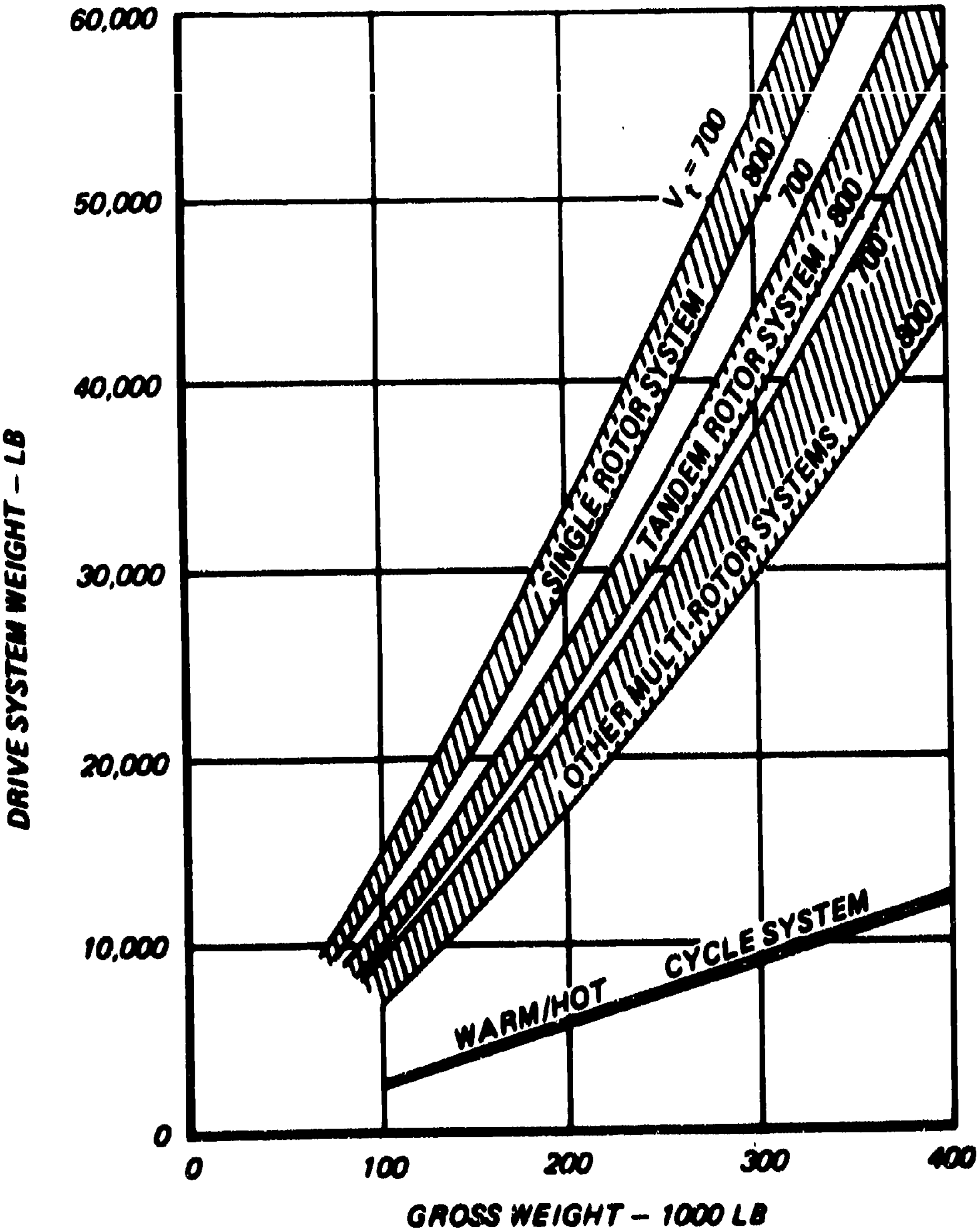
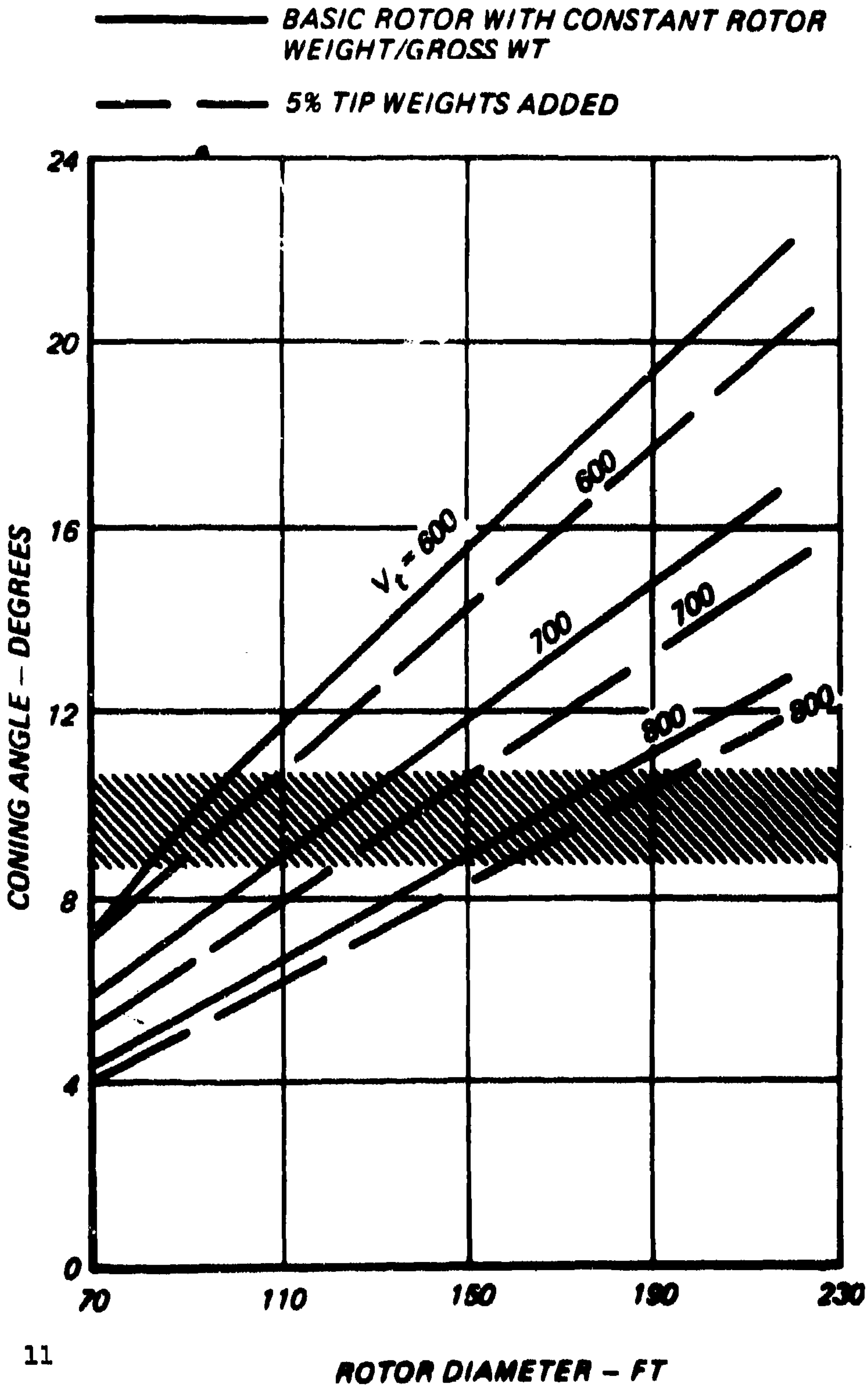


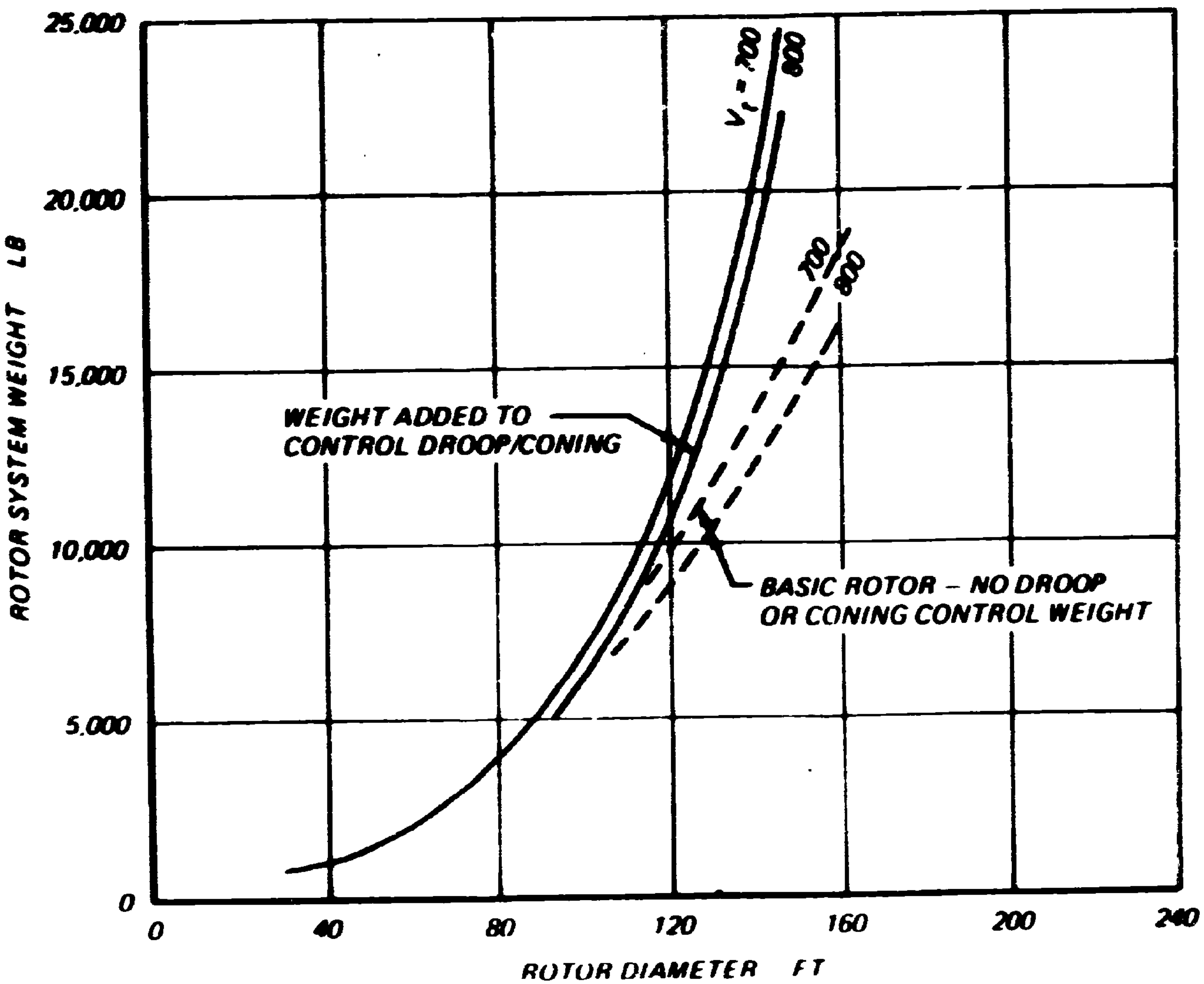
FIGURE 10

Total Drive System Weight
(Disc Loading 10 PSF)



BASIC ROTOR WITH CONSTANT ROTOR WEIGHT/GROSS WT
5% TIP WEIGHTS ADDED
CONING ANGLE – DEGREES
ROTOR DIAMETER – FT
24
20
16
12
8
4
0
70
110
150
190
230
V_t = 600
600
700
700
800
800

FIGURE 11



Rotor System Weight (Disc Loading = 10
$C_T/\sigma = .115$)

FIGURE 12

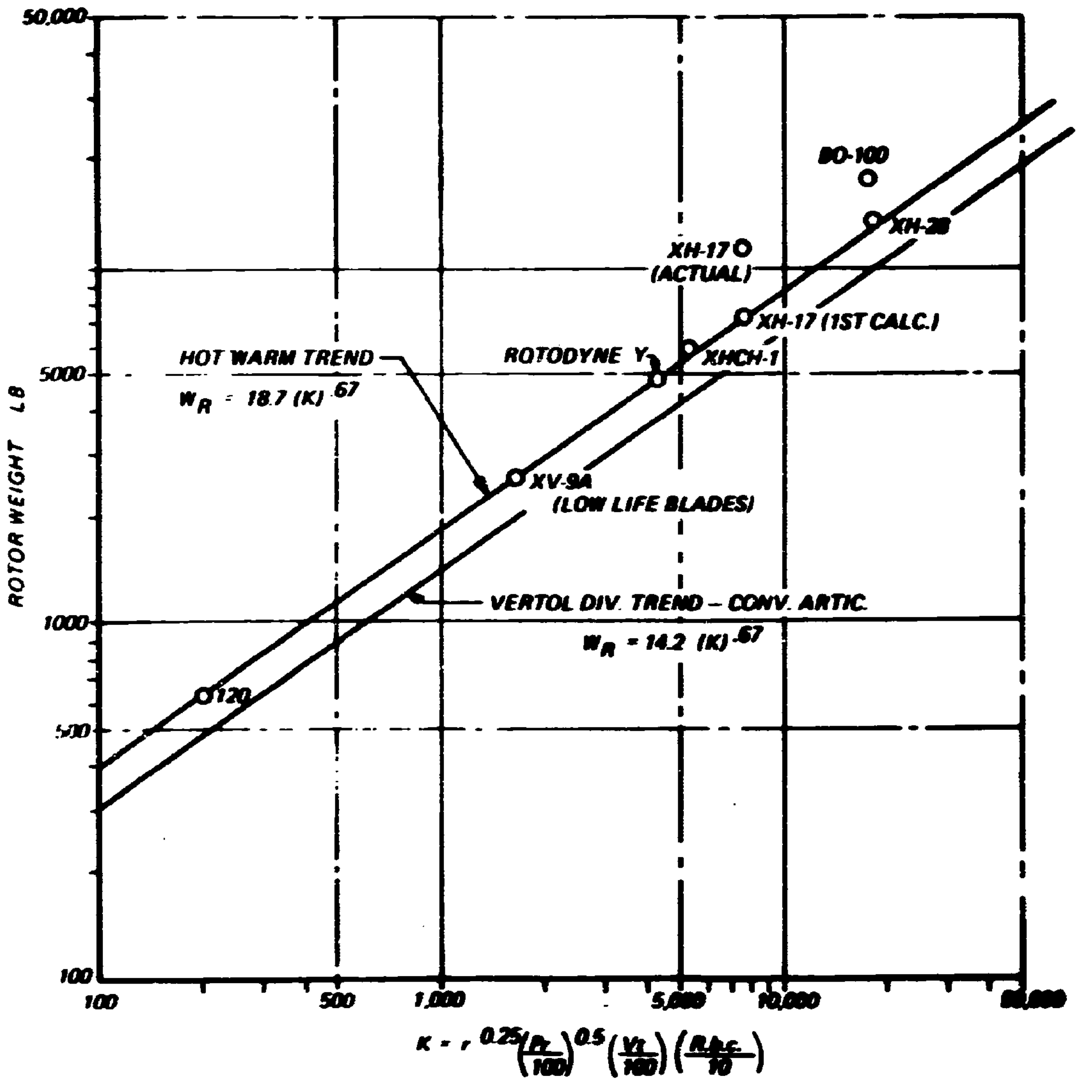
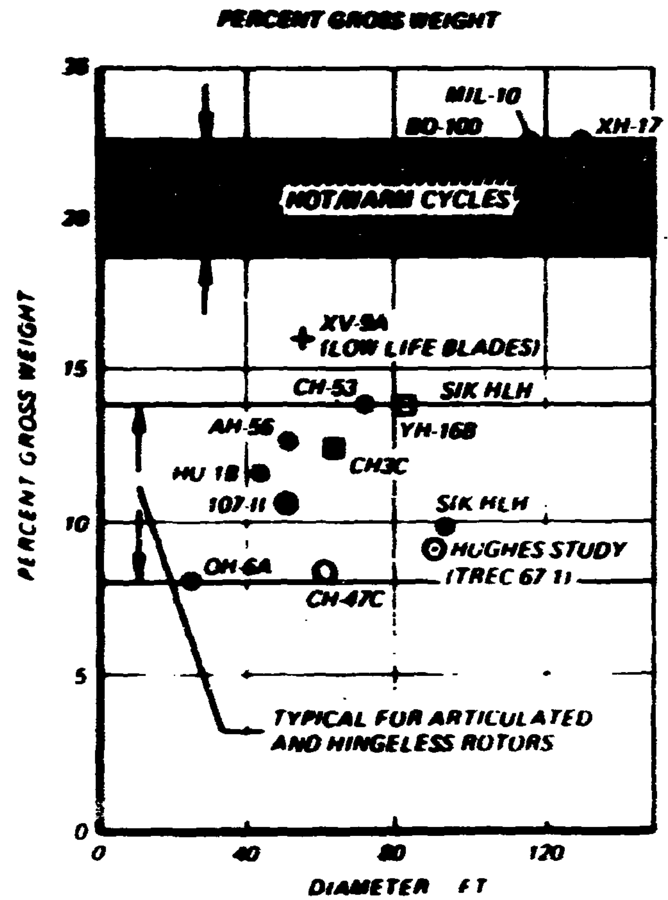
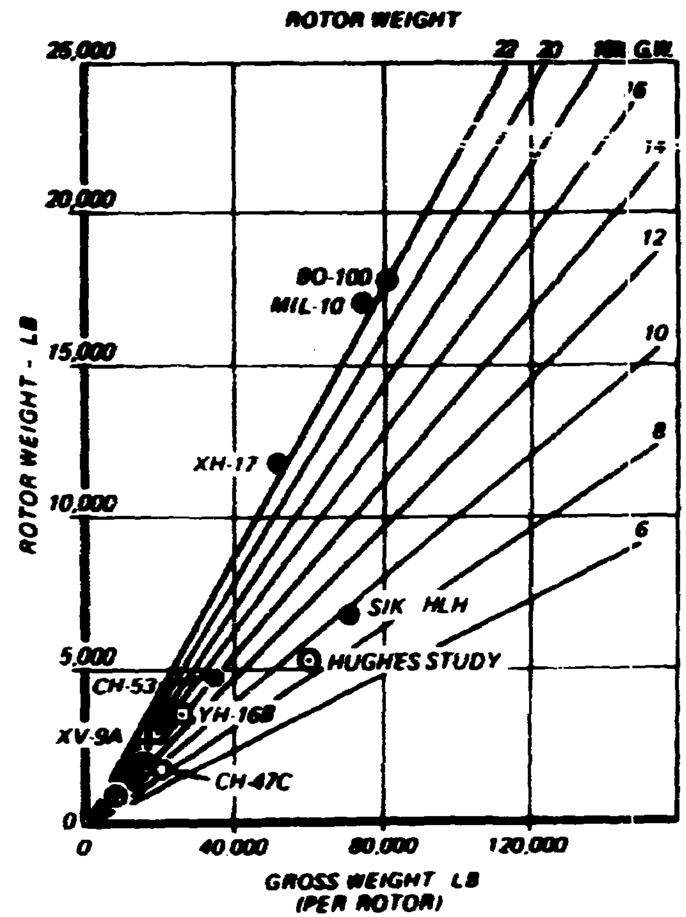


FIGURE 13 **Rotor Weight Trend**

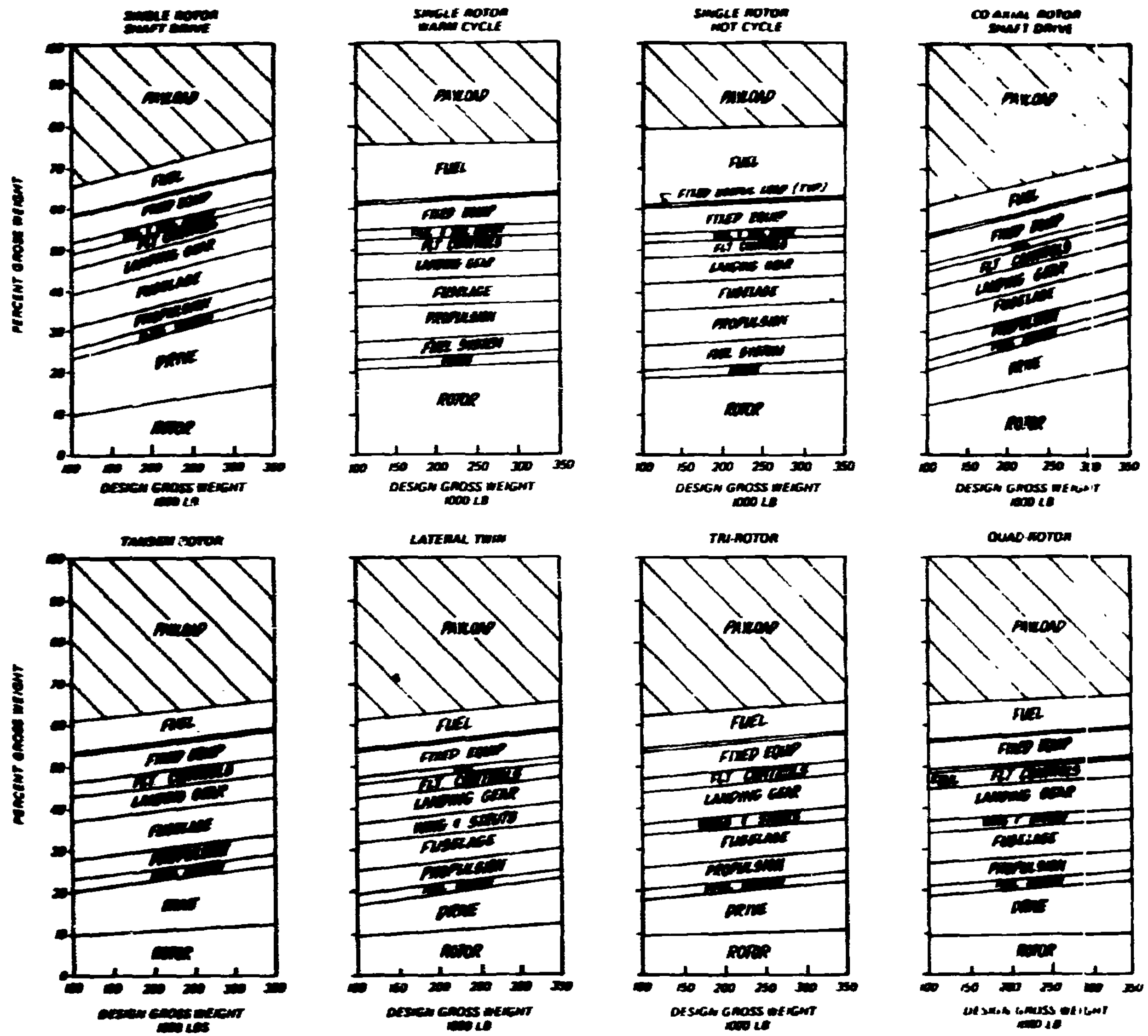




Rotor Weight (Gr Wt for HOGE 4000–95°)

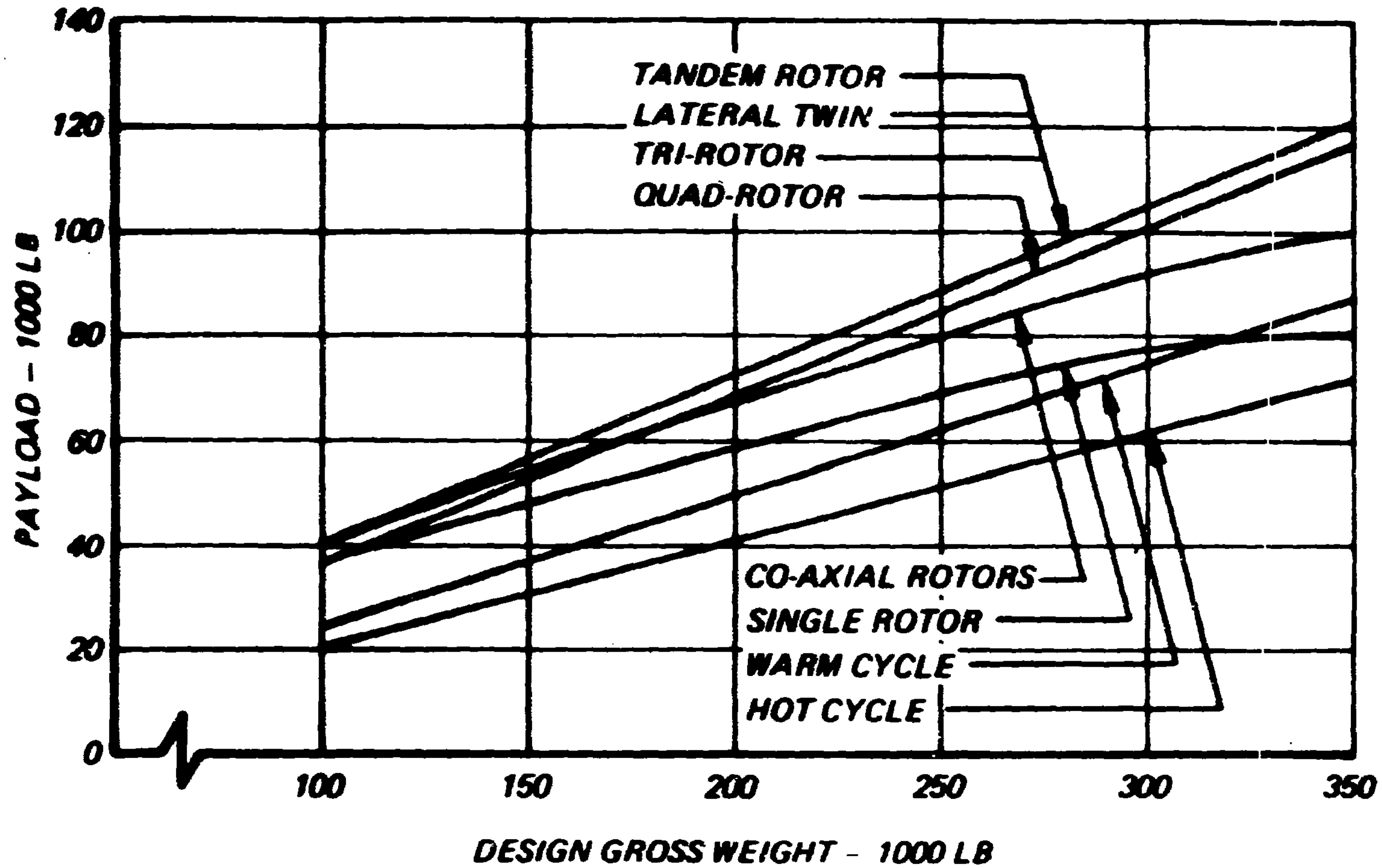
FIGURE 14

200-11



Component Weights Breakdown.

FIGURE 15



Payload Capability

FIGURE 16

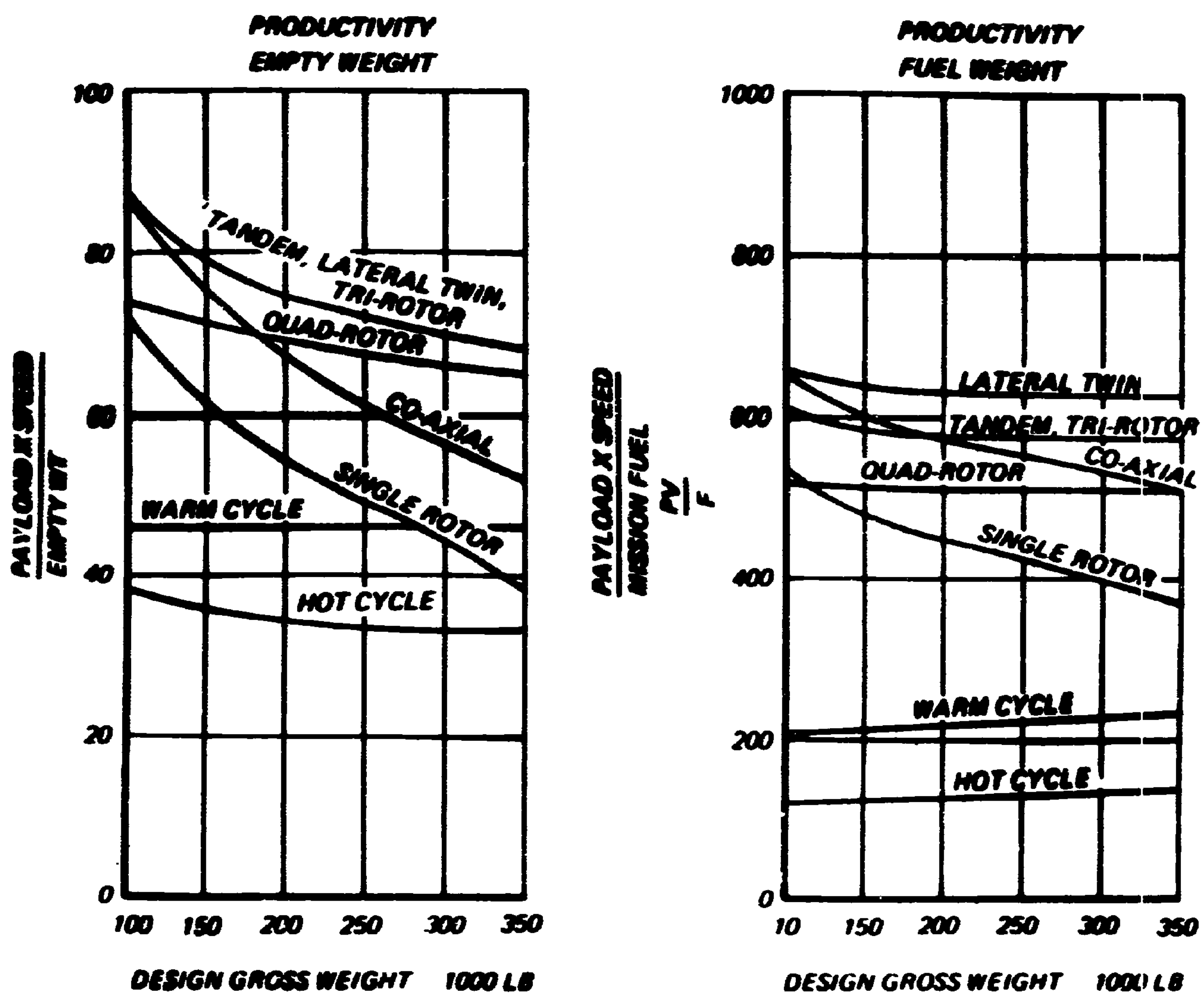
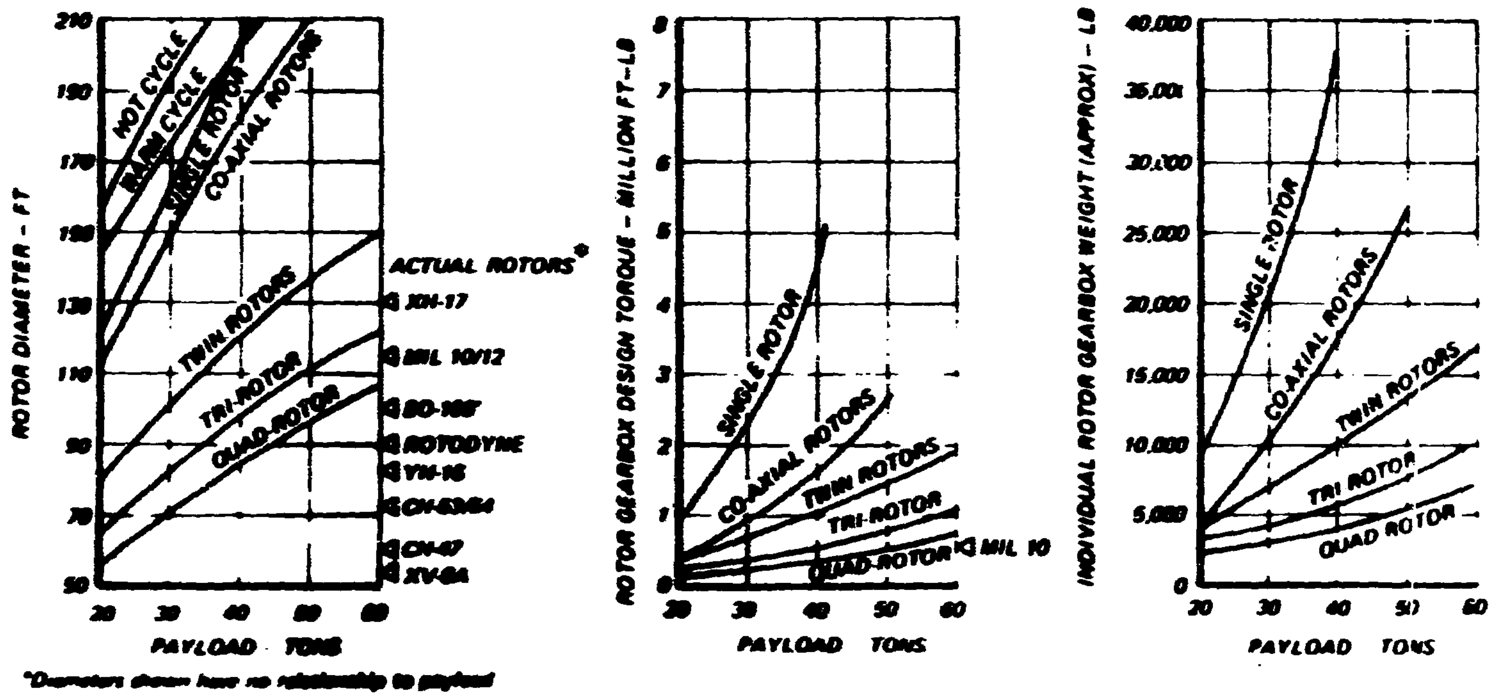


FIGURE 17 **Relative Productivity**



Summary of Sizing Risks

FIGURE 18

APPENDIX M
SIZE EFFECTS

Presentation by Mr. Edward S. Carter, Sikorsky

This discussion will briefly cover some of the work we have been doing on the question of size effects and the impact it will have on some of the basic criteria we generally accept. The 50-60 ton payload VHL aircraft that we have been talking about today is a mighty big vehicle, and we have not developed the very large building blocks in this country as our Soviet friends have; we have an awful lot to learn. Figure 1 illustrates some of the areas we have been thinking about.

TAKE OFF CRITERIA:

The first is take-off criteria. What effect does the very large rotor have on the hover power margin requirements that have been receiving so much attention lately? Next, what is the effect of very large airframes with very big loads operating in confined areas on handling quality requirements? Then, what is the effect on our dynamic load criteria that goes along with this? Also, what is the effect on the frequency trends when we get into much larger aircraft? Blade frequencies and the airframe frequencies are going to change, and they are going to have a big impact on our rotor design. And then, finally noise.

Let us consider the take-off criteria first. It basically breaks down into the question of the power margin that you require to get in and out of a confined area which we feel is a function of the blade radius that you are dealing with. By this, we mean the requirement that we heard cited yesterday of 500 feet-per-minute vertical rate of climb imposed on top of an OGE requirement. We are not talking about the temperature and altitude requirement. Each customer must tell us where they are going to operate the machine, and then we must design it for the temperature and altitude requirement. But once you have accepted that - what power margin do you require to operate from a confined area? Most of the thinking that has generated current criteria has been derived from experience with the UH-1's and other smaller aircraft. This experience involves 40 to 50 foot rotors operating in confined areas where the obstructions can be from 50 to 75 feet high. There will be a very different situation when we start operating 160 to 200 foot rotors over the same trees. The trees are not going to change. We doubt very much if a VHLH will ever hover out-of-ground effect, and we think we see indications of this trend in both the CH-54 and the Mil 10 experience. We have talked to pilots who have flown the Mil 10 and there is no question that it is an in-ground effect vehicle. In fact, it is used in a STOL mode a good deal of the time. The experience with the CH-54 also bears on the problem. It is the closest thing we have ever produced to a flat-rated aircraft. It should have come closer to meeting the Army's requirements for power margin than any other helicopter in the field today. But what happened in Vietnam? Instead of flying it as designed for GW from 38 to 42 thousand pounds, we are flying it at 47 thousand pounds, and you find that the power margin available for any given temperature and altitude, instead of meeting OGE or 500 feet-per-minute over OGE requirement, is coming out a little more like 10% below OGE. Figure 2 illustrates the experience that we have been recording in service in terms of the spectrum of power margins as the pilots are flying the CH-54. The abscissa is the amount of power available relative to what it would take to hover OGE. By far, the majority of the work being done in S.E.A. is so far in-ground effect that they can operate with close to a minus 10% power margin. To be sure we are not at all certain that there is not a little cheating on transmission ratings going on; but the fact remains that if the aircraft was constrained to operate at the recently proposed criteria, it would be limited to gross weights way below the majority of current utilization.

The message here is that usage will rise to meet the IGE capability and we must recognize this if we are to achieve a balanced design. If we build a 50-ton crane with a 500 foot-per-minute vertical rate of climb over OGE, we are going to find that we have a 70 ton crane by the time that it is in service. However, this is apparently a very disputable point. There is the evidence I have just cited, but there are plenty of people in the procuring service that do not agree. So, I would like to urge this group to support this <u>testing</u> to find out what these power margin requirements really are in confined areas, and how they are affected by size. Very little of this has been done, particularly, with very large helicopters. We ought to take the CH-54 and get some real side-by-side flight testing with a smaller helicopter to pin down the effect of size. Compare the CH-54 to the UH-1 and find out what it takes in the way of power margin to operate in the same confined areas.

HANDLING QUALITIES:

Handling qualities characteristics affected by size are principally, the maneuver requirement and the effect of pilot location on the kinesthetic environment we are putting him in. With reference to maneuver requirements, we again learn interesting things from Mil 10 pilot experience. From all reports it is a very, sluggish machine. The Soviets have sacrificed no payload to provide a fast or snappy response. Apparently pilots just accept the fact that you must handle such a large aircraft with kid gloves. The situation is somewhat analogous to trying to dock the Queen Mary. Nobody expects to hot-rod the Queen Mary into New York Harbor like a kid with an outboard. Yet, at the moment, our handling qualities criteria, and our demands for maneuverability show no real recognition of size effect except for the control power consideration. Even such things as engine response apparently is very, very sluggish on the Mil 10. While it is true this is one of the things that western pilots thought was too marginal, the fact remains that with large size aircraft there is a question of whether you can afford the luxury of all of the maneuverability that we have been accustomed to on smaller aircraft.

Figure 3 indicates another way that size effect impacts control criteria. It illustrates what has happened to pilot location as we have made larger aircraft. Virtually all of our current VTOL types depend upon inclining the lifting vector in order to obtain horizontal acceleration. This means anytime that you want to accelerate you must tip the aircraft and, when you tip the aircraft, since the pilot is removed from the center of gravity, you are going to put a significant acceleration on him. The H-34, which was always a very nice pilot's airplane, had the pilot located quite high relative to the axis of rotation and just barely ahead of it. In the H-37, he was relatively high and well forward of the c.g. On the CH-54, the pilot is beginning to get well forward of the axis of rotation and we are reaching the point where, when we make a yaw correction, we have to anticipate a good deal of lateral acceleration, unless, as we suspect to be the case, the pilots are ingenious enough to rotate the aircraft around themselves instead of around the c.g. However, for crane missions, the pilot must rotate around the c.g. to avoid translating the hoist point.

Consider, now, the VHLH. The pilot is going to be at least twice as far out ahead of the c.g. or hoist point. So again we have an area we need to study. We are not saying it is insurmountable nor that it is a spacing problem but it is something that we need to do basic homework on. How are these pilots going to fly this aircraft? Are they going to rotate around the c.g. or are they going to rotate around themselves; and if around the c.g., what impact does this have on optimum control response.

A special aspect of this problem which also occurs in large fixed-wing aircraft is rotation and landing gear impact during the landing flare. With the pilot so far out ahead of the rotation axis he may well have a tendency to fly the rear end of the airplane into the ground. As a matter of fact, we have already seen this phenomena on the CH-53 where we found that Service Pilots were frequently landing with a rate of descent on the landing gear at touchdown significantly in excess of what they thought they were imposing, simply because as they flared the aircraft, they flared it around themselves and tended to fly the rear end into the ground.

This is just one of those natural trends that we have all seen evidence of. The point is, that when we talk about building an airplane that is perhaps four times the gross weight of anything we have flown today, we must identify these areas for some basic homework. We think this would be a very good problem for the 6-degree of freedom similator at NASA Ames.

DYNAMIC LOADS

I will not talk too much about the question of dynamic loads. It is pretty evident that it goes without saying that, if we are going to change our maneuverability requirements, this will have an impact on the load factors and angular accelerations. There is always the question of gust response and how this is going to be altered in much larger aircraft. We are getting to the point where the size of the rotor relative to the size of the gust can be significantly different, thus another area that needs homework. On the other hand, frequency trends are a special concern in very large rotary wing aircraft. We are concerned with what is going to happen as the 1 per rev excitation from the rotor begins to come pretty close to the airframe first bending mode. The general trends are indicated on the right side of Figure 4. It is even possible that the VHLH aircraft will be large enough so that you might get to the point where the aircraft airframe modes could be low enough to couple with the in-plane degrees of freedom of the rotor ("Air Resonance").

However, an even more critical problem may be the human response resonances. We know that the human being is very unhappy if he is exposed to 2 to 5 cps, and as you push up the rotor diameter (holding the tip speed), you lower the rpm, and as you lower the rpm you get your frequencies down where we do not want them. The right side of Figure 4 shows the trends which result. With a six-bladed rotor, on a 40,000 lb aircraft we are getting N per rev frequencies around 17 cycles. The pilots seem to like this as a range for excitation frequency. We notice that we can get away with a good deal more g's at 17 cycles than they did at 10 cycles in the old H-19. But as we go to bigger sizes, even the six-bladed rotor is going to drop down into the below 10 cps range. On the left side of Figure 4 is a relative annoyance versus frequency plot as a constant G level. So the message is that we will be forced to more and more blades to keep the frequencies up, and we must do the homework required to be able to use large numbers of blades efficiently.

NOISE

I have only one other brief subject to touch on, and that is the question of noise. Several people asked yesterday whether anyone has done any thinking on what the noise problem will be. We have done a little bit, as illustrated in Figure 5. One inevitably feels that with power, comes noise. You can change the spectrum a good bit - and you can put weight in to reduce the noise problem but, inevitably, if we are putting in a lot more power, there is going to be more noise for a given level of acoustic control. Figure 5 plots against power the PNdB level of several helicopters to give a little feel for the problem. The H-3, at a distance of 200 ft, is at about 103 PNdB now. The H-54 and H-53 are around 114. We think that we should be able to keep on a trend line that is below the goal for the quiet H-3 program. So with the technology at our disposal today, we think we should be able to get down to the lower line, especially if we can take advantage of the change in the frequency distribution that you should have in the VHLH. Even with this, however, we must estimate that your VHLH will probably be in the 117 PNdB ball park.

SIZE EFFECTS

T.O. CRITERIA -

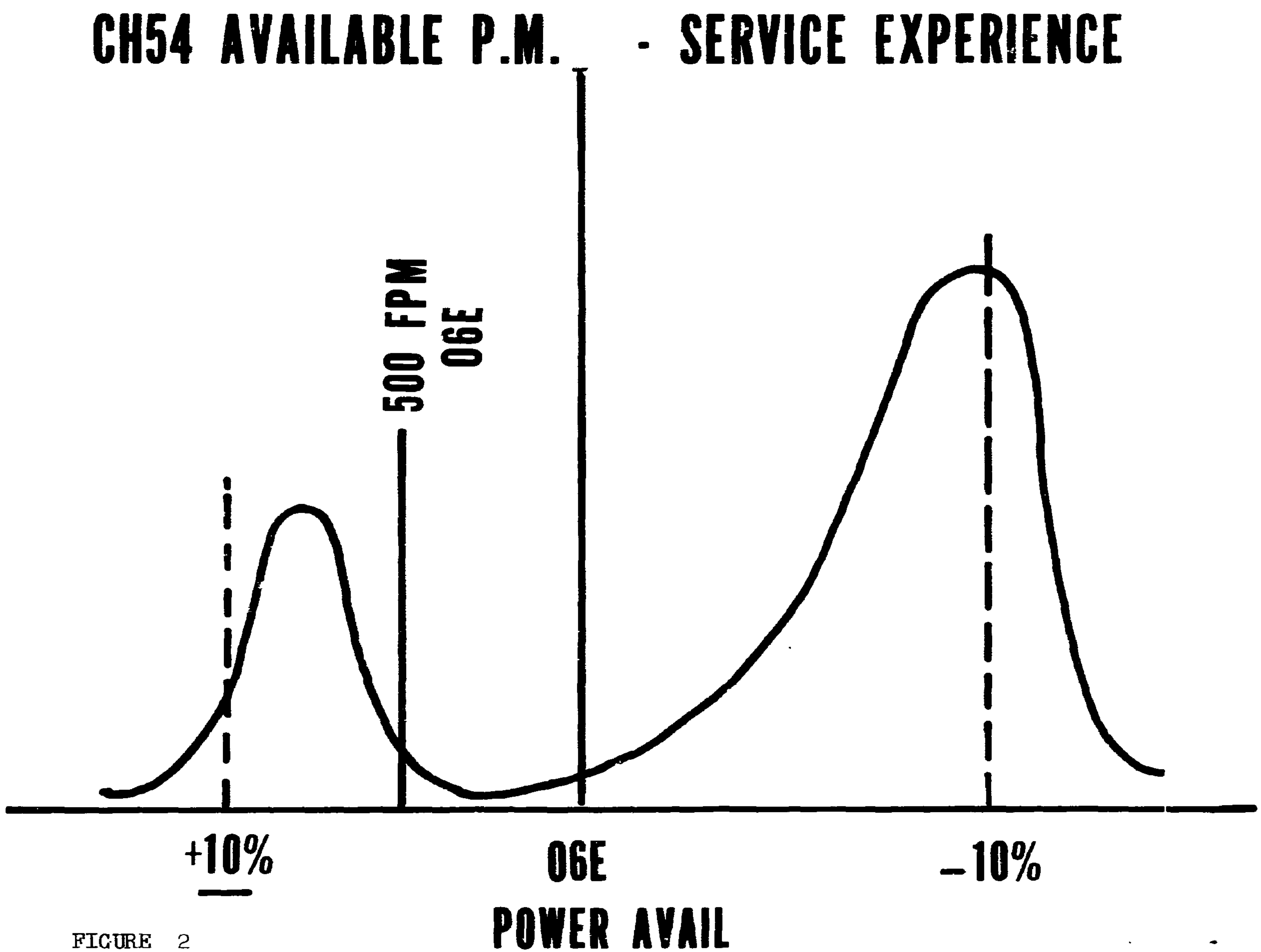
HANDLING QUALITIES

DYNAMIC LOADS

FREQUENCY TRENDS
(AND ROTOR DESIGN)

NOISE

FIGURE 1



CH54 AVAILABLE P.M. - SERVICE EXPERIENCE
500 FPM
06E
+10%
06E
-10%
POWER AVAIL

FIGURE 2

PILOT LOCATION vs AXIS OF ROTATION

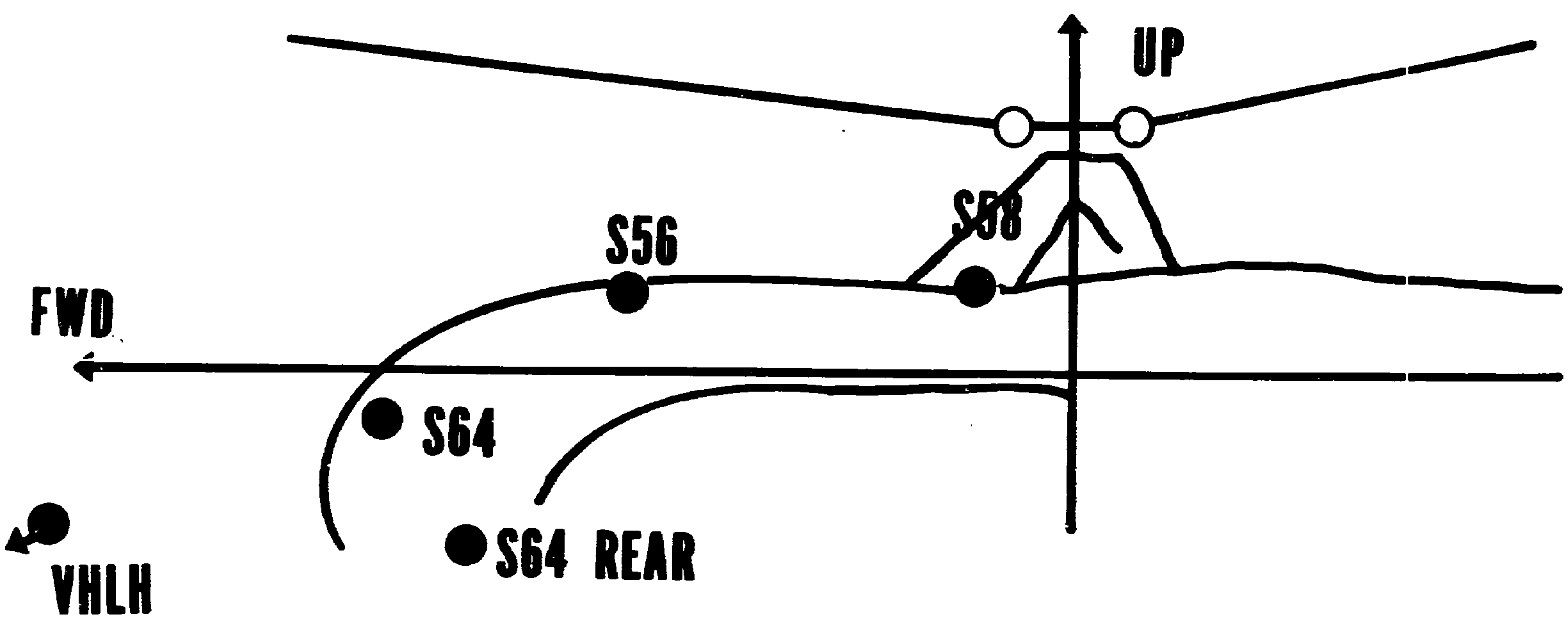


FIGURE 3

EFFECT OF SIZE ON VIB. FREQ

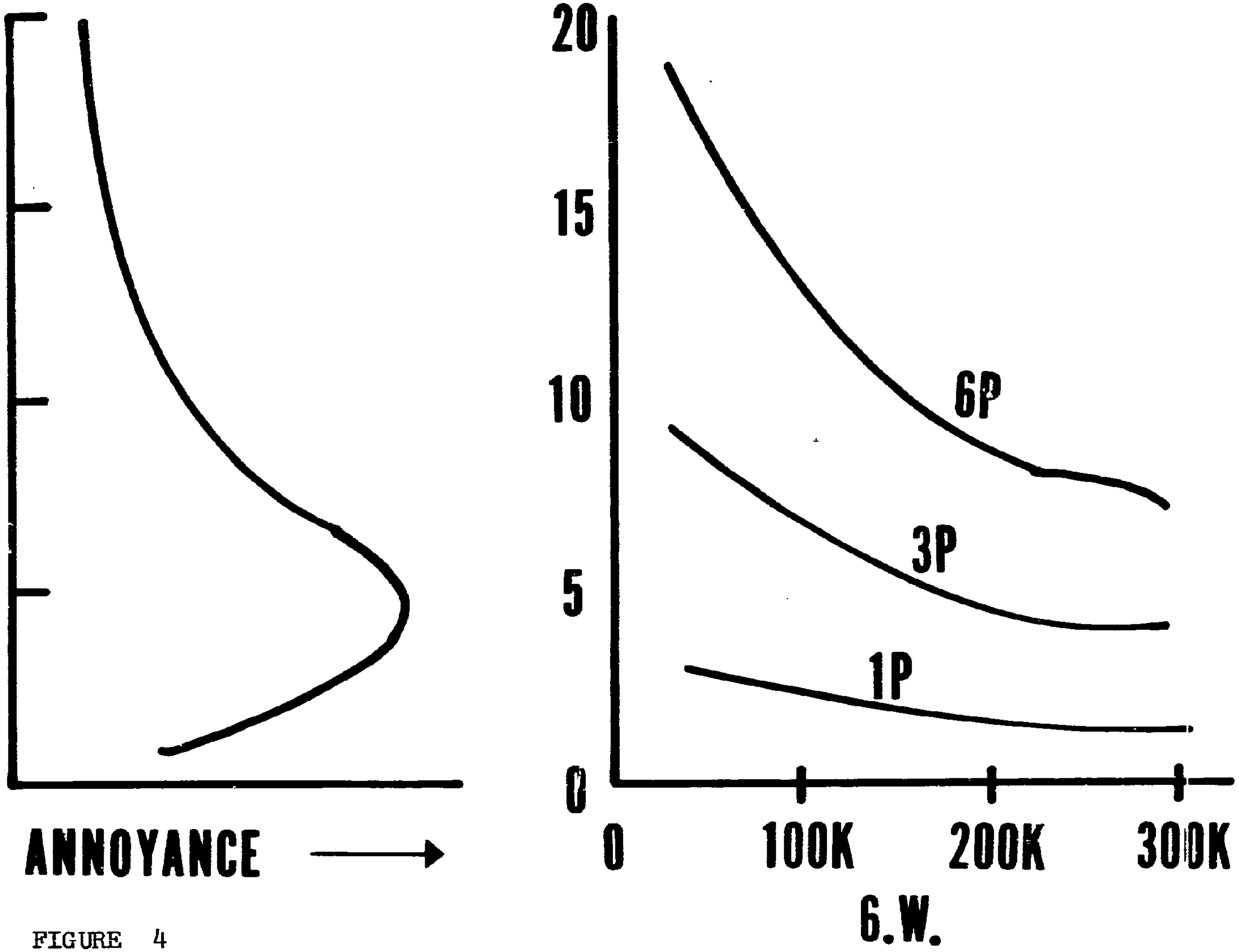
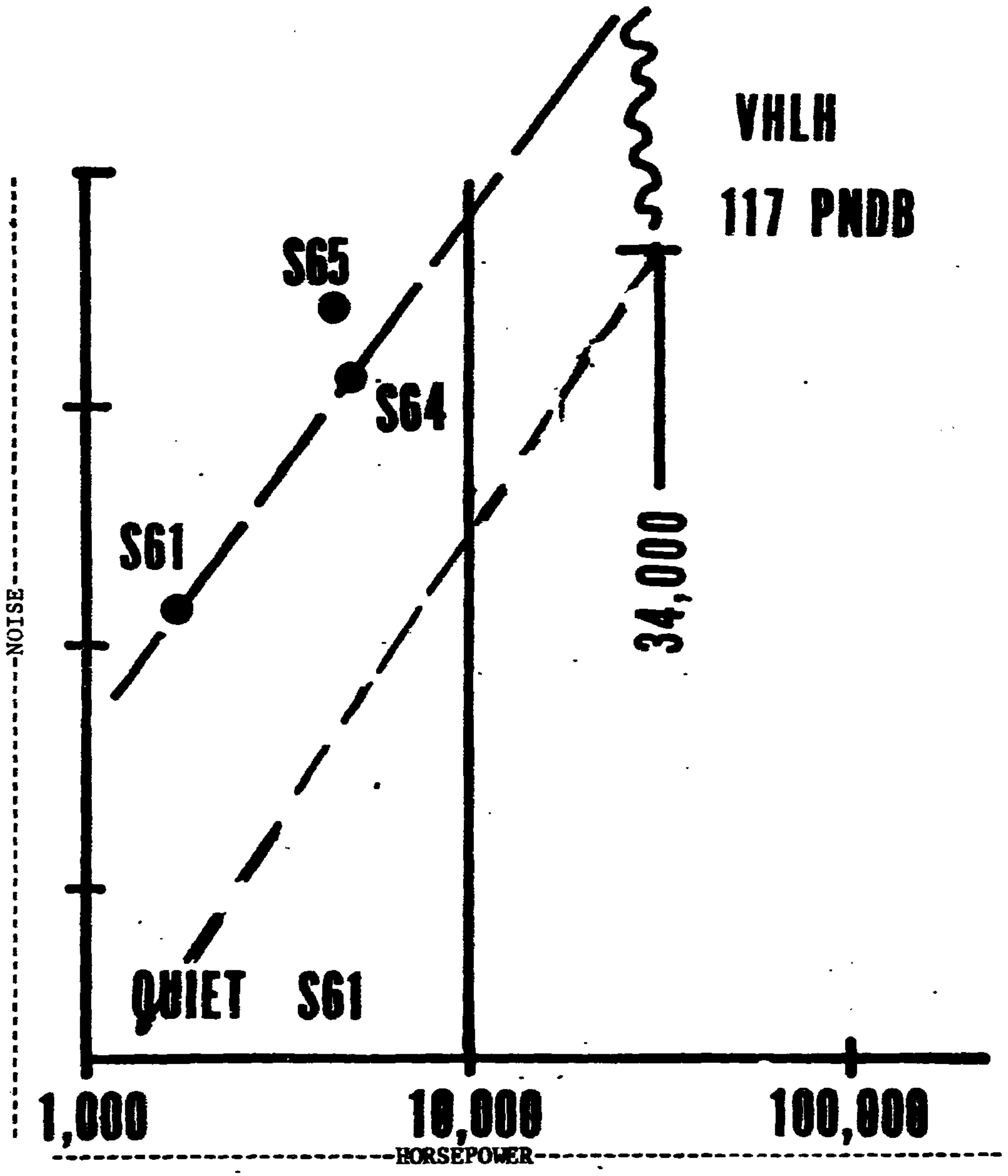


FIGURE 4



VHLH
117 PNDB
S65
S64
S61
34,000
QUIET S61
NOISE
1,000
10,000
100,000
HORSEPOWER

FIGURE 5

APPENDIX N
FRG ADVANCEMENTS RELATED TO THE VHLH

Presentation by Herr K. Pfleiderer - Messerschmidt-Bölkow

First of all I want to confirm all the data given by Mr. Schneider (Boeing Vertol) and Mr. Amer (Hughes Tool Company) about the efficiencies of the tip-driven helicopter rotor systems. There is good agreement between their results and those obtained at Bölkow on various test rigs with cold and warm cycle systems. There is, however, one specific point on which I disagree with both gentlemen; and this is the weight of the rotor system. We define it as the weight of the hub, the blades and the stationary and rotating gas ducts. These estimates seem to me to be too pessimistic in the cast of Mr. Schneider's and too optimistic in the case of Mr. Amer's presentation.

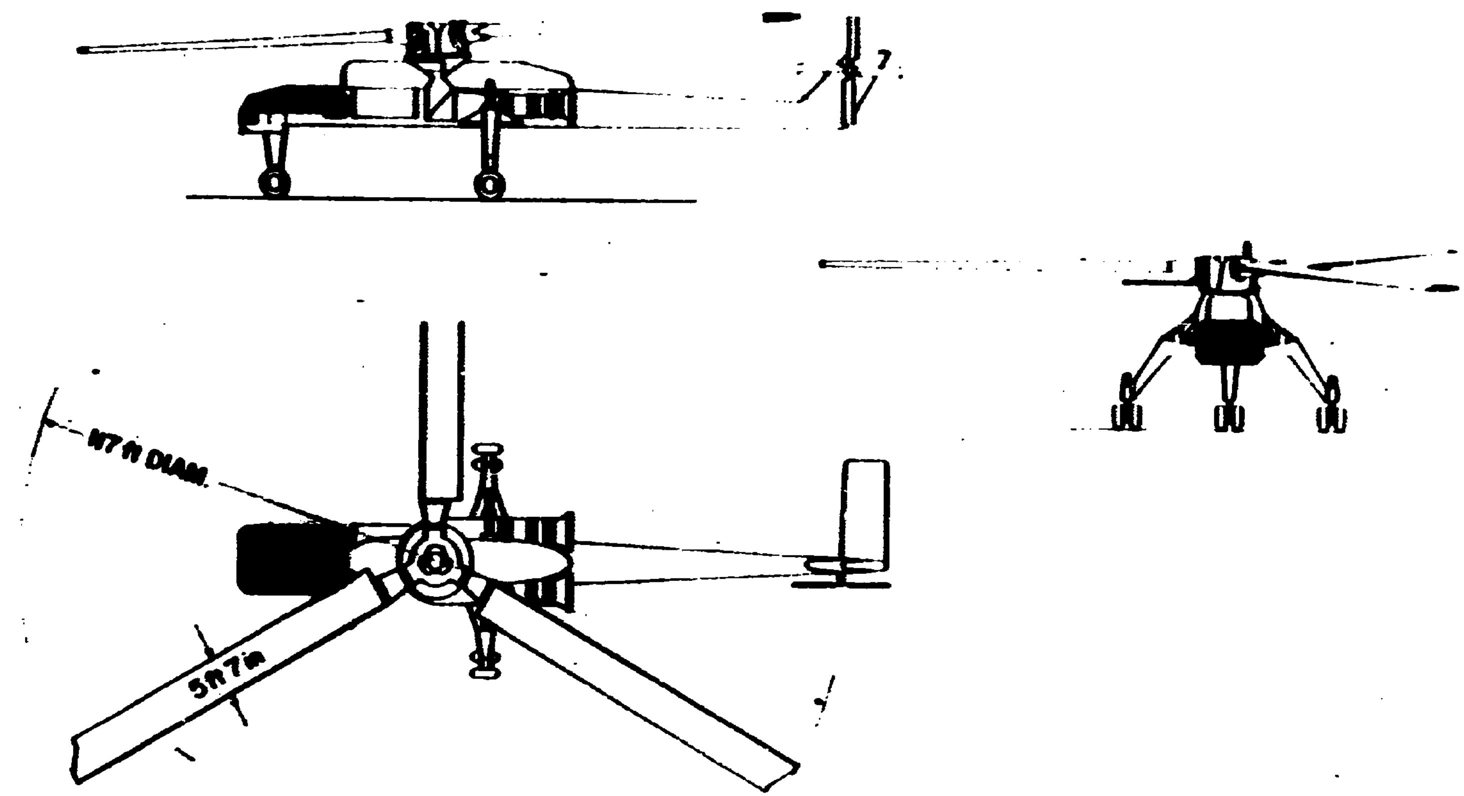
The weight estimates given by Mr. Schneider are partly incorporating Bölkow data, with the restriction that this data applies to steel rotor hubs and not to those "titanium rotor hubs," that have been designed in the meantime. The current opinion, in our company, is that rotor system weights of 15-16 percent of all-up-weight can be reached. Figure 1 shows a drawing of a 50-ton crane helicopter project for which all of our research work has been done. Figure 2 shows a review of the total program at Bölkow on tip-driven helicopters in the past years. We started with a 4m diameter cold cycle test rig and ended up with a 31m diameter warm cycle rotor test bed on which most of the quoted efficiency data were measured.

During all of our research, we found an interesting relationship between blade-chord, bypass ratio, and pressure ratio. Looking for the optimum drive efficiency (that is, for minimum fuel consumption per net thrust in hovering flight), it was found, that a bypass ratio of M = 2 fulfills this objective. In Figure 3, calculated efficiency data are plotted against the chord of the rotor blade for 3 and 4 blade rotors. In this figure there is included a line for the mean lift coefficient of the blade of $\bar{C}_L = 0.6$. At the beginning of the meeting, as I remember, we heard from Dr. McCormick that he suggests a $\bar{C}_L$ of 0.45 as a suitable value. Using either of these values one ends up with the same chord of the blade as one would when dimensioning for the minimum fuel consumption criterion. That means if you talk about 3 or 4 blade rotor systems, there is no difference in blade chord between shaft and tip-driven designs, and, of course, none between hot or cold cycle systems. These results can also be seen from Figure 4 which shows the optimum blade chord for various tip speeds of tip-driven systems as a function of bypass ratio. The indicated line for $\bar{C}_L = 0.6$ is influencing the chord of the blade up to the optimum bypass ratio of M = 0.2. The only design parameter left is the thickness of the blade, and I'll show you later that by chordwise unbalanced blades you have no duct area problems when using conventional thickness ratios for the blades. This is valid for hot or warm cycles but the warm cycle offers many advantages in lifetime, weight, simplicity, etc. Figure 5 shows "gas temperature and gas pressure" as a function of bypass ratio. As you have no duct area problem regarding the $\bar{C}_L$ requirement, you now of course will choose the warm cycle with only 500-600°F and a pressure ratio in the order of 2.1. At this gas temperature you can employ currently available materials and technologies. You also - and this is a very important point - can use existing fan engines. This will bring the advantage of being able to utilize future engine developments with higher turbine inlet temperatures and compression ratios, resulting in lower fuel comsumption; all this without requiring new blade construction technologies. These would have to be developed if you took the gases from simple gas generators with bypass mixing, that is, hot cycle for 1400°F and more.

To save blade weight and to get a maximum gas duct area, we looked at the flutter behavior of chordwise unbalanced blades. We ended up with blades that have their center of gravity at 32 percent chord. Test results and analytical studies showed that no danger of flutter exists throughout the flight regime. This is achieved by the high torsional stiffness of those blades. From Figure 6 you can see these blades have 3-times the torsional frequency ratio than conventional rotor blades. Figure 7 shows a cross section of such blade. The load carrying structure is of elliptical shape and, therefore, a near optimum for withstanding the internal pressure, which is 100-times higher than the external aerodynamic loads. This ratio would increase to twice this value when using pure gas generators and, therefore, would cause severe structural and blade weight problems. Figures 8 and 9 show those two blade designs out of a dozen; we considered the best to be an aluminum fiberglass composite blade. Two filament wound fiberglass tubes are bonded together to form an elliptic structure. HRH-Honeycomb webs are bonded to this inner structure before the outer aluminum skin is fixed to it, also by a bonding process. A similar structure was proposed for use on the SST-airplanes. The second blade design is a stainless steel sandwich construction made with the AVCORAMIC process. To prevent high thermal stresses between the inner and outer skin, the inner one is insulated by a non-structural high-temperature (Polyimide) fiberglass sandwich against the warm gas flow. Figure 10 shows our favorite hub design. We designed five different hubs for weight estimates and comparisons. Using steel as structural hub material we ended up with 20-21 percent rotor weight of the total weight of the helicopter. That is the number Mr. Schneider has already mentioned. By using titanium, we reduced it down to 16 percent.

Let me now give you an interesting result concerning the fuselage weight of such a big helicopter. Meeting MIL 8501, a long tailboom and a tail rotor outside the main rotor circle is lighter than a short one and you can save about 50 to 60 percent the tail-boom weight. You also need less horsepower for your control and the helicopter has a better directional stability. Comparing the total weight of the fuselage of shaft and tip-driven helicopters, we obtained an over all difference of about 15 percent, the fuselage of the tip-driven one being lighter.

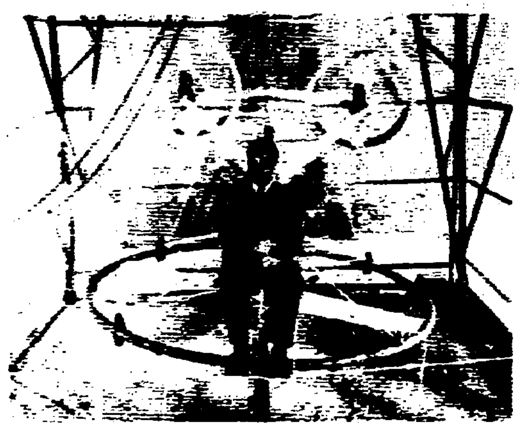
Let me conclude with the problem of cockpit vibration. I think the comment, that the low frequency vibration of a large 3-bladed rotor acting on the pilot can be a limitation on such helicopters and may necessitate some passive or active damping devices, is important. This problem should be studied immediately. A cockpit vibration isolation can only be used for transport helicopters. But I think that we should take into account that some of these helicopters will sometimes have passenger compartments fixed to the fuselage. Therefore, all possible ways of fighting the problem should be thoroughly investigated.



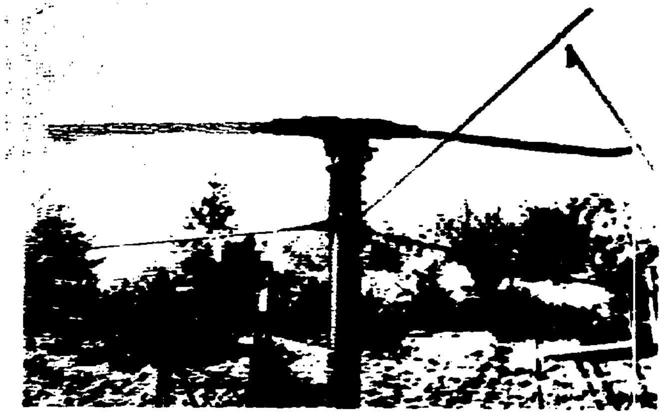
BO-50 to CRANE HELICOPTER PROJECT

FIGURE 1

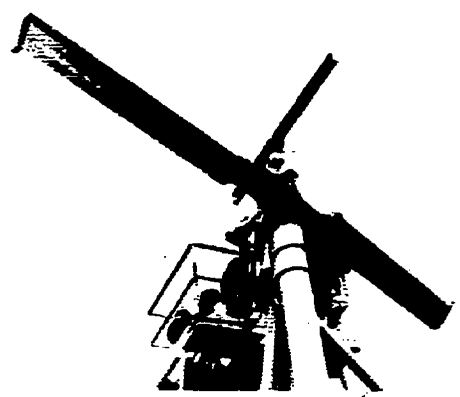
208-14



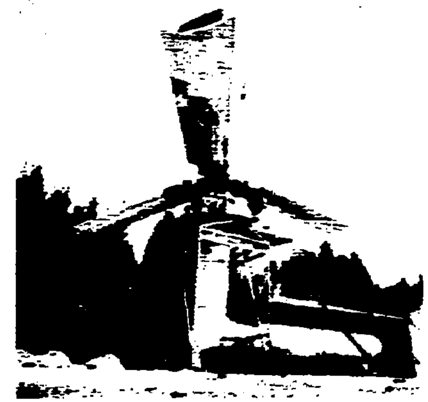
FLYING JEEP



4m - KALTGASREAKTIONSROTOR



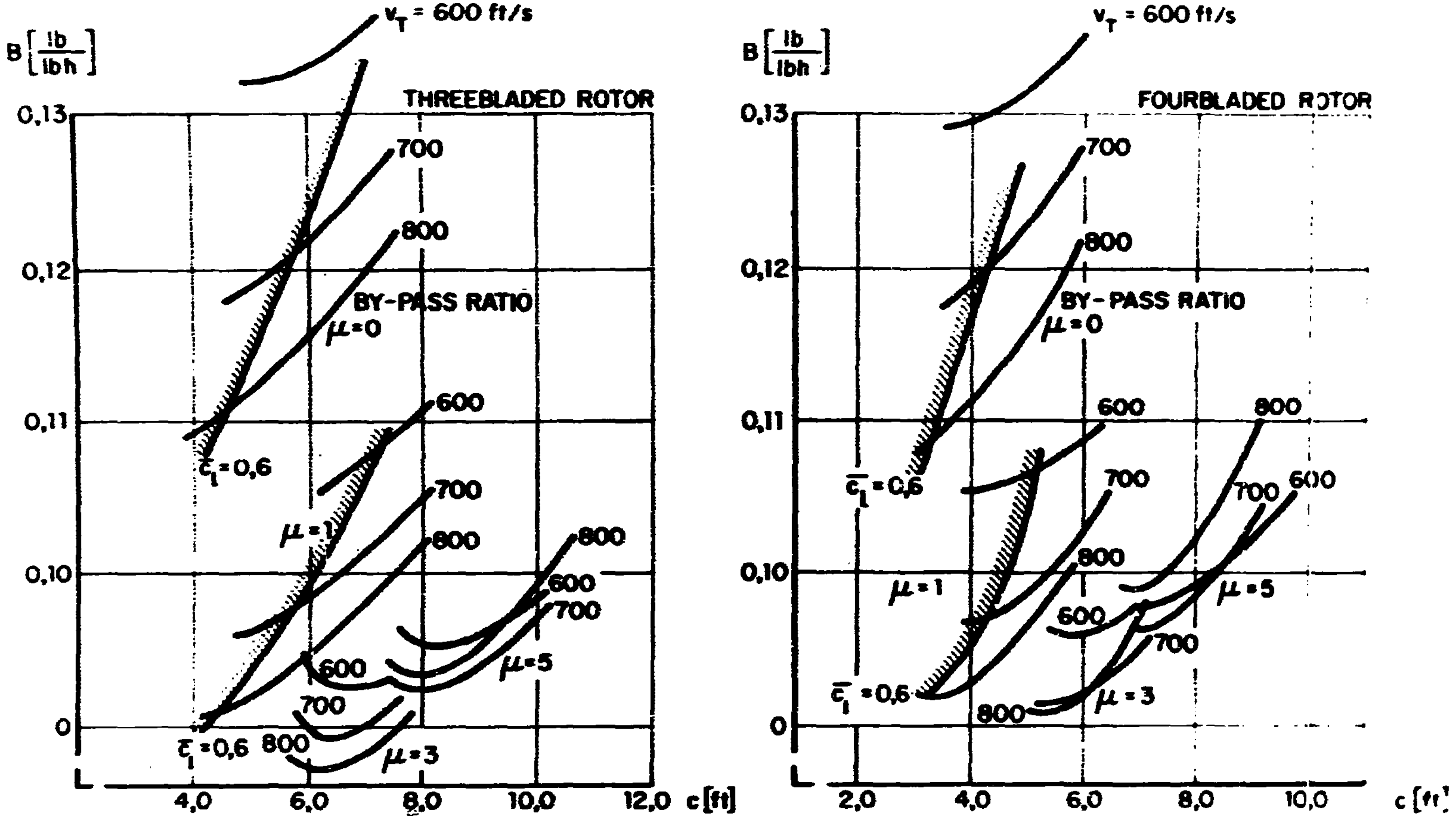
8m - MISCHGASREAKTIONSROTOR



31m - MISCHGASREAKTIONSROTCR

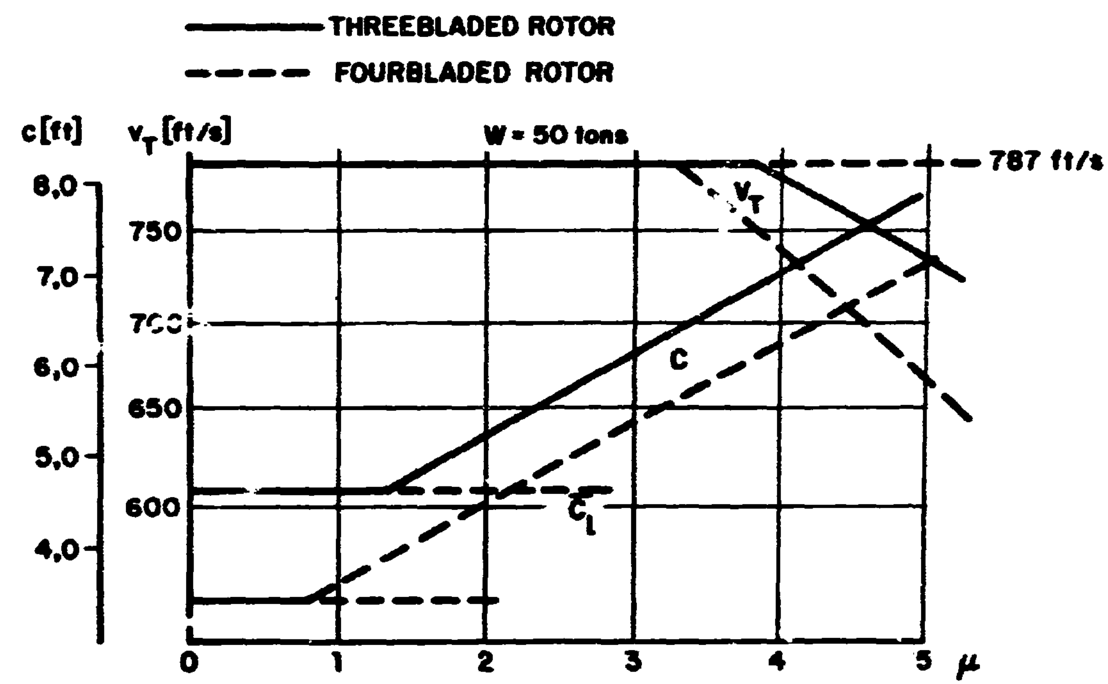
REAKTIONSGETRIEBENE ROTOREN
DER BÖLKOW GMBH

FIGURE 2



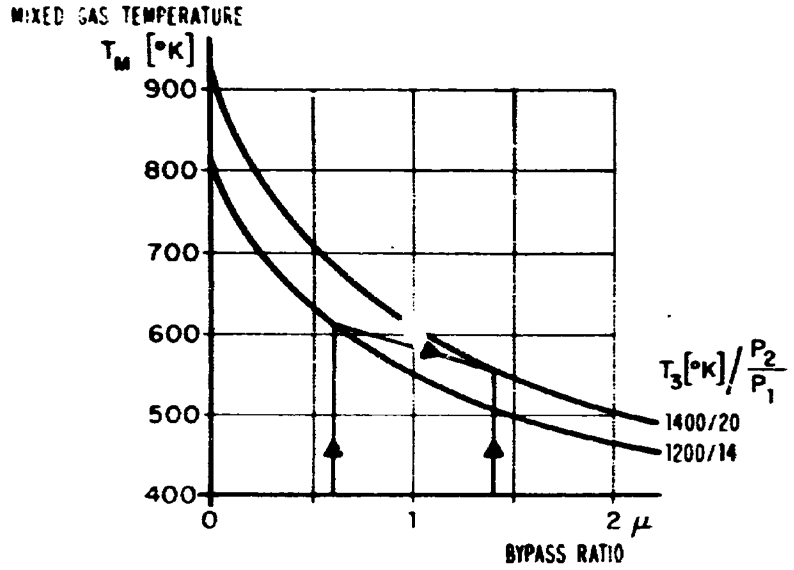
HOURLY FUEL CONSUMPTION PER NET THRUST

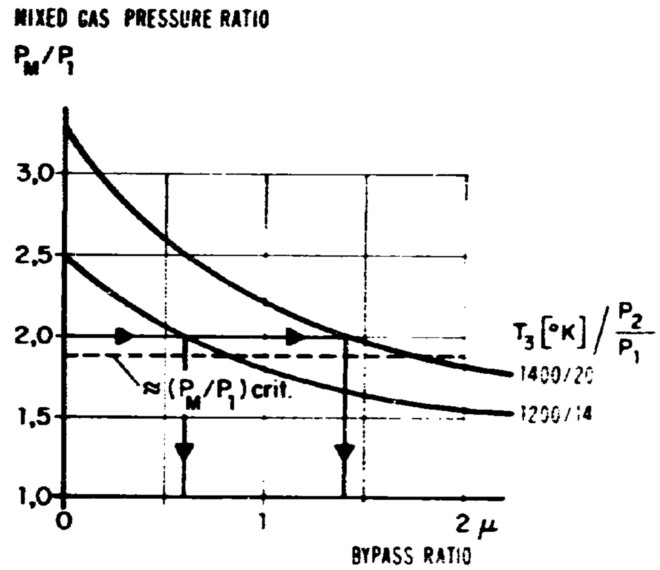
FIGURE 3



OPTIMUM BLADE CHORD AND TIP SPEED

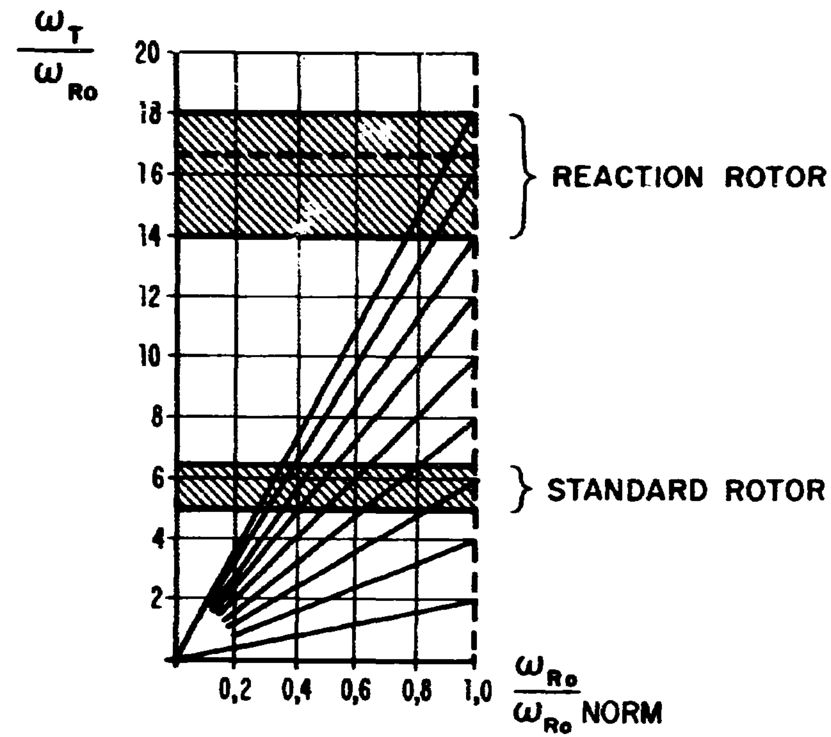
FIGURE 4





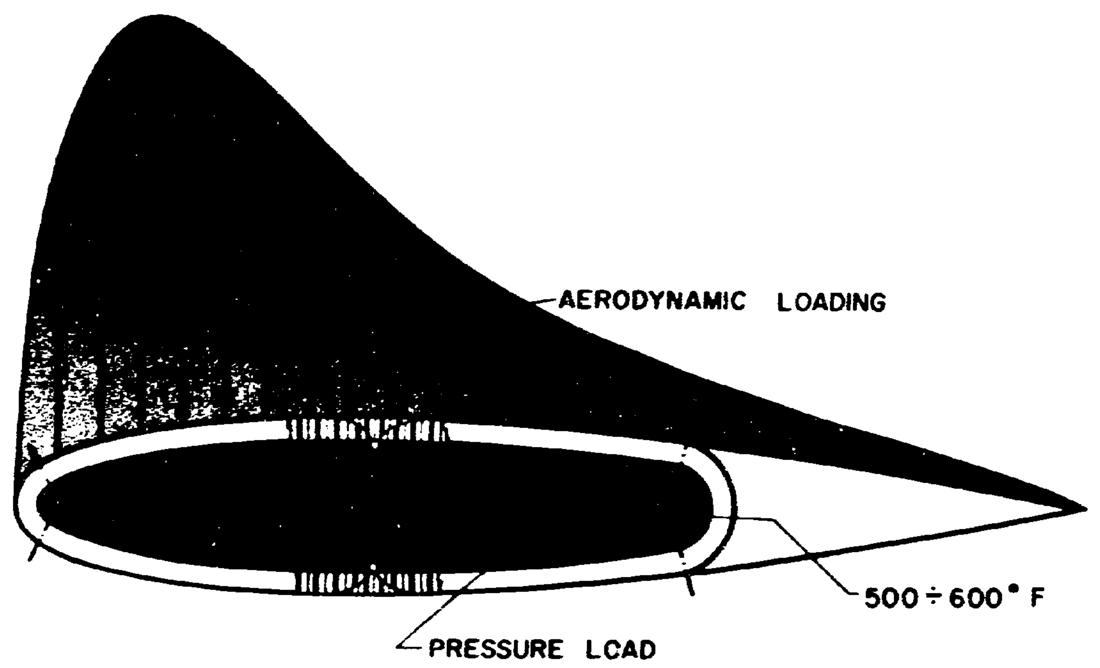
GAS TEMPERATURE AND PRESSURE RATIO

FIGURE 5



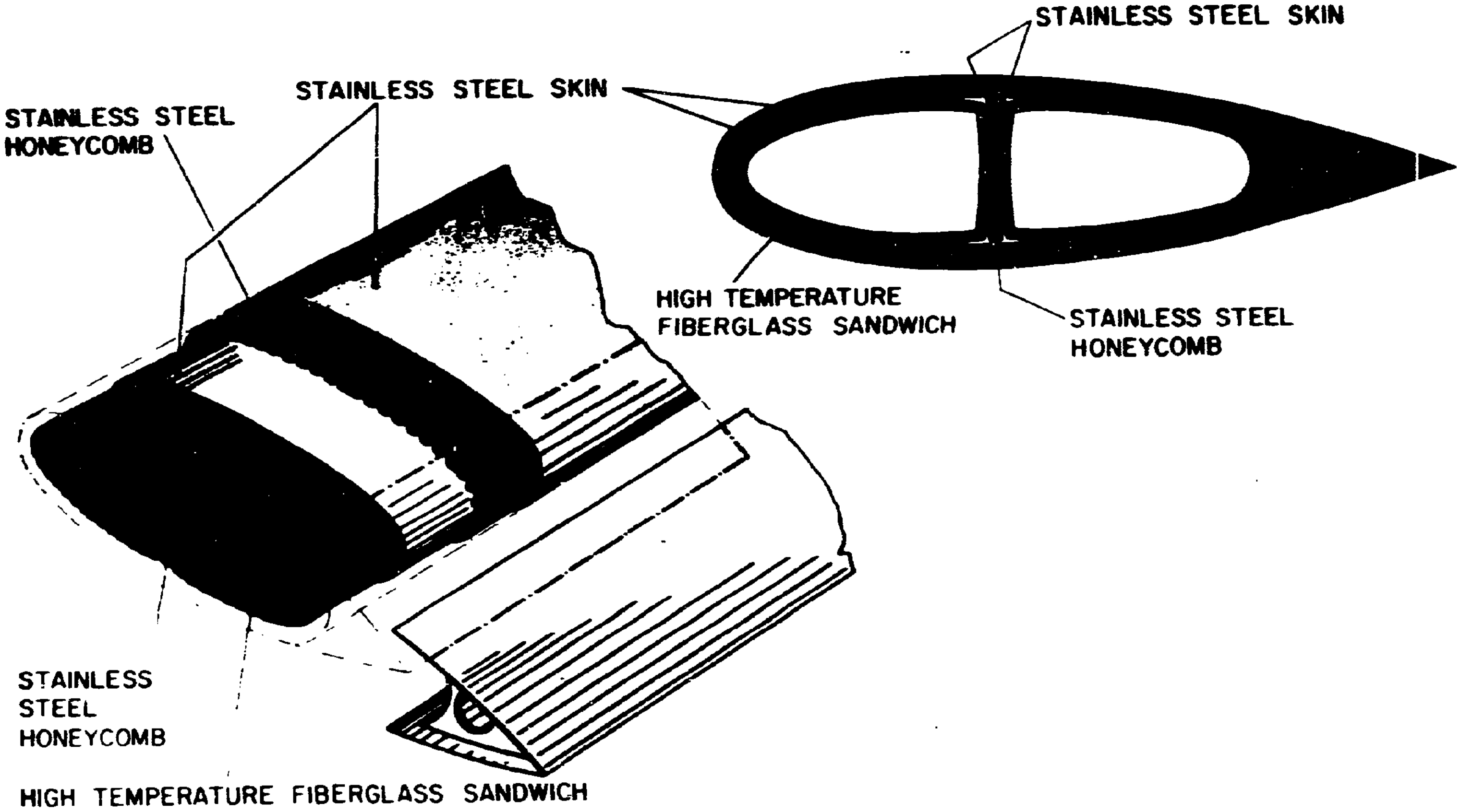
NATURAL TORSIONAL FREQUENCY OF ROTORBLADES

FIGURE 6



ROTOR BLADE LOADS

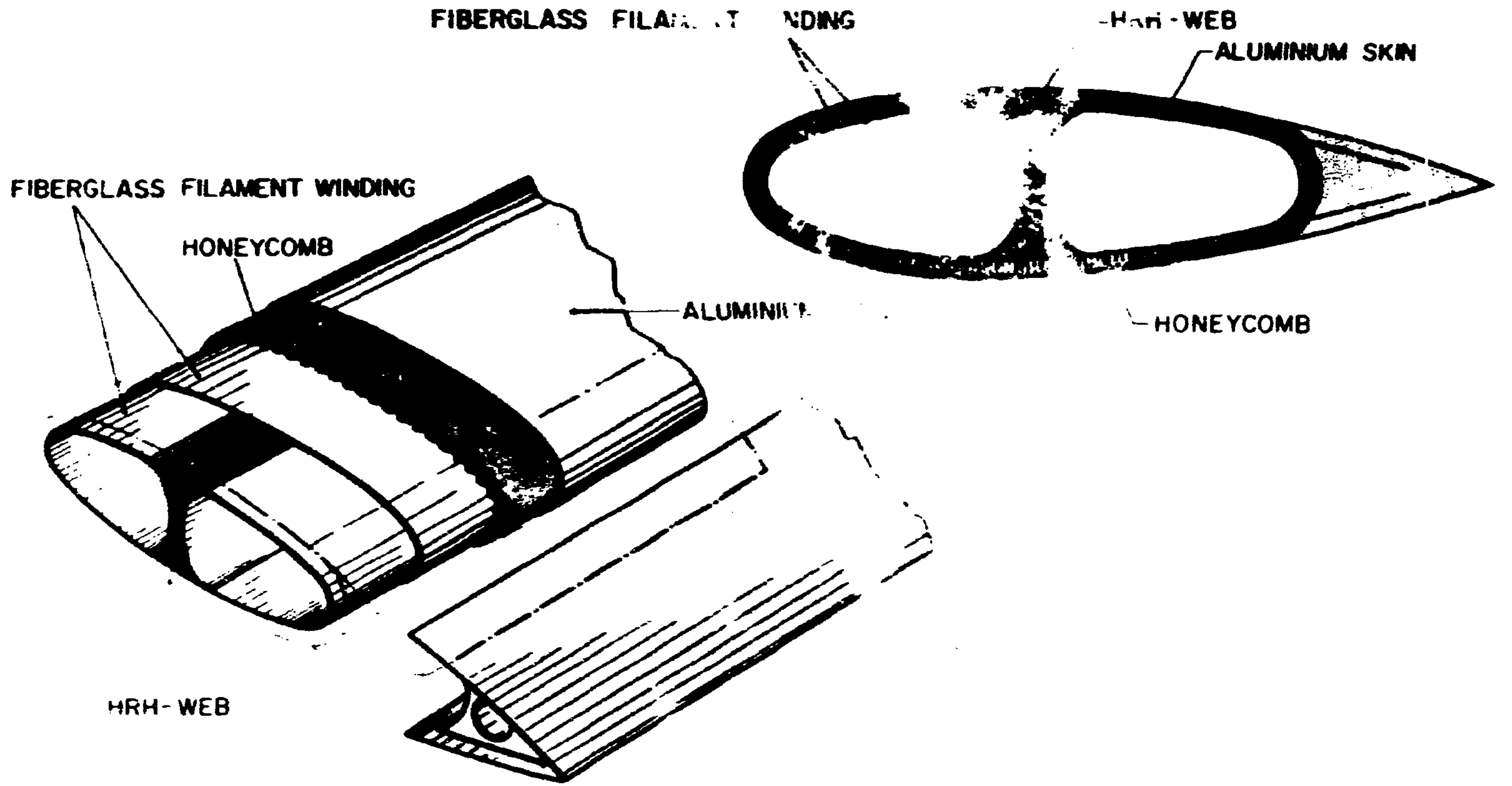
FIGURE 7



STAINLESS STEEL BRAZED SANDWICH

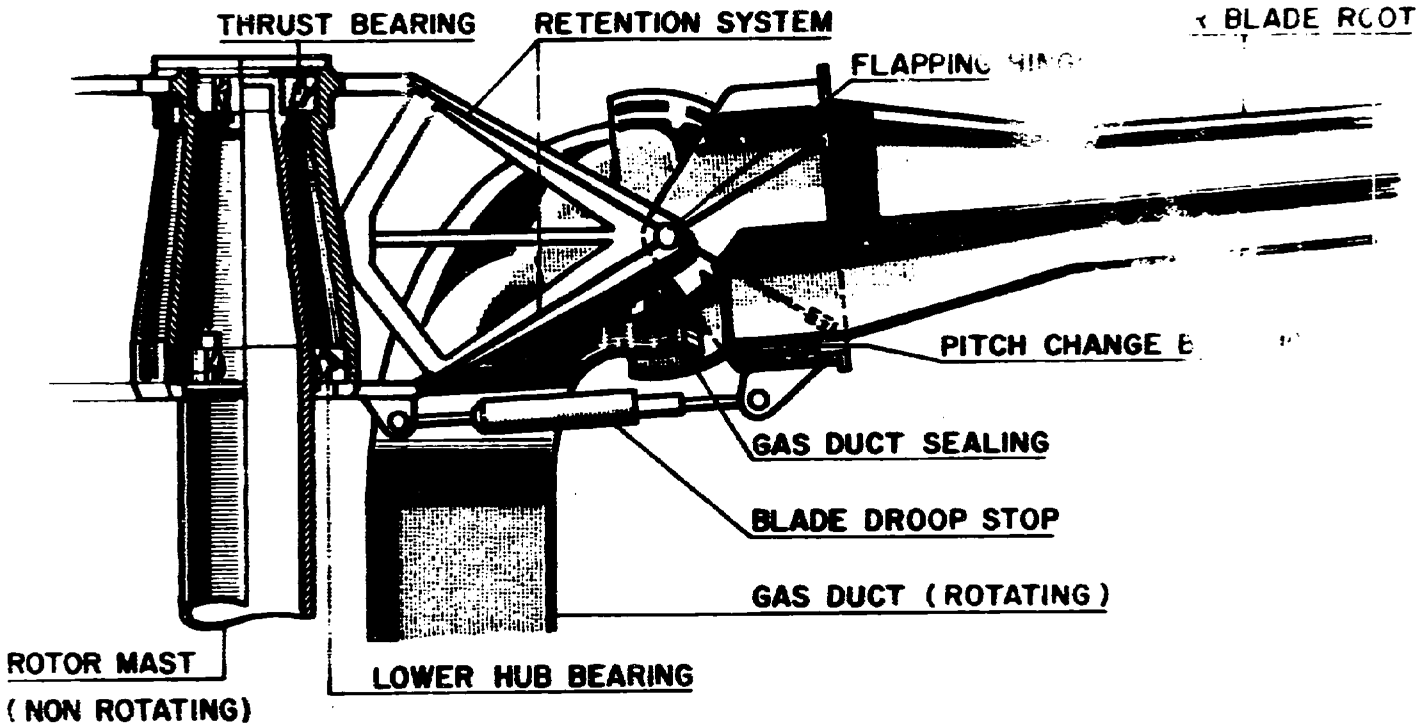
FIGURE 8

200-14



ALUMINIUM - FIBERGLASS - COMPOSITE

FIGURE 9



SEMI-RIGID ROTORHUB - SYSTEM

FIGURE 10

UNCLASSIFIED
Security Classification

DOCUMENT CONTROL DATA - R & D

(Security classification of title, body of abstract and indexing annotation must be entered when the overall report is classified)

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13. ABSTRACT

Vol III covers the Academic and Industrial Presentations at the subject AHWG. Given below is the abstract for Vol I, the Basic Report (AD 862 287L):

An Ad Hoc Working Group (AHWG) was convened by the US Army Advanced Materiel Concepts Agency (AMCA) to consider the technical feasibility of developing a very heavy aerial lift vehicle to be operational by 1990. The mission spectrum for such a vehicle was assumed similar to the proposed QMR for the heavy-lift helicopter except for the payload requirement which was increased in these considerations to approximately 50-60 tons. Shaft-driven and tip-driven helicopter concepts were given major consideration. Power-plant and transmission development and problems relating to subsystems, such as load handling and control, were treated. Vertical take-off and landing (VTOL) concepts that were not high disc loading were discussed. Short take-off and landing (STOL) was not considered.

14. KEY WORDS	LINK A		LINK B		LINK C	
	ROLE	WT	ROLE	WT	ROLE	WT
Aircraft						
Helicopter						
Vertical take-off and landing (VTOL)						
Aerial Heavy Lift						
Shaft-Driven						
Tip-Driven						
Dirigible						